REPORT GDC-DCL67-009
NASA CONTRACT 8-18118

# FINAL REPORT
# LARGE SPACE STRUCTURE EXPERIMENTS FOR AAP

VOLUME I • SUMMARY



GENERAL DYNAMICS
*Convair Division*

[redacted]

NASA CONTRACT 8-18118

# FINAL REPORT
# LARGE SPACE STRUCTURE EXPERIMENTS FOR AAP

## VOLUME I ✦ SUMMARY

30 November 1967

Prepared for
ADVANCED SYSTEMS OFFICE
MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

Prepared by
CONVAIR DIVISION OF GENERAL DYNAMICS
San Diego, California

NASA Project Manager
W. T. Carey
Code R-AS-VO, NASA-MSFC
Huntsville, Alabama 35812

Convair Project Manager
J. R. Hunter
581-60, P. O. Box 1128
San Diego, California 92112

# FOREWORD

This report presents the results of a study of "Large Space Structures Experiments for AAP" conducted by the Convair division of General Dynamics for the Marshall Space Flight Center, NASA. The study was performed during the interval 15 September 1966 to 15 September 1967 at a level of approximately $275,000 under contract number NAS 8-18118. The final report is published in five volumes:

Volume I Technical Summary

This volume summarizes the results of the entire study.

Volume II Analysis and Evaluation of Space Structure Concepts

This volume presents the results of the analysis of the 40 space structure concepts investigated during the first half of the study.

Volume III Crossed H Interferometer for Long Wave Radio Astronomy

This volume contains the design details of the Crossed H Interferometer, which was one of the three concepts selected at mid-term for detailed analysis.

Volume IV Focusing X-ray Telescope

This volume contains the design details of the X-ray Telescope, which was one of the three concepts selected at mid-term for detailed analysis.

Volume V 100-Foot Parabolic Antenna

This volume contains the design details of the Parabolic Antenna, which was one of the three concepts selected at mid-term for detailed analysis.

## ACKNOWLEDGEMENT

Orientation and technical guidance have been provided by the NASA-COR, Mr. W.T. Carey of MSFC. The completeness of the study has been enhanced in the area of crew systems through the coordination of Mr. N. Belasco of MSC. Many companies, listed in Volume II, have submitted concepts for analysis and evaluation. Major contributors to the study included the following Convair personnel:

| | |
|---|---|
| Mr. J. A. Fager | Volume V |
| Dr. M. C. Nilson | Astronomy |
| Mr. B. P. Swett | Volume IV |
| Mr. G. E. Taylor | Volume III |

Convair would like to acknowledge the unselfish and invaluable assistance, at no cost to NASA or Convair, of the following members of the scientific community:

Dr. C. Stuart Bowyer,
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American Science and Engineering, Inc.
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Dr. G. Richard Huguenin,
Harvard Space Radio Laboratory
Dr. George R. Lebo,
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University of Michigan
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Massachusetts Institute of Technology
Dr. N. Frank Six, Jr.,
University of Western Kentucky
Dr. Robert G. Stone,
Goddard Space Flight Center
Dr. P. Vandenbout,
Columbia University
Dr. John R. Waters,
American Science and Engineering, Inc.
Mr. T. E. Wing,
Columbia University

Most of the consultants met with Dr. Matt C. Nilson of Convair in the initial phases of study, to help extract design constraints and goals appropriate for the time period under consideration. The majority of the members of the astronomy community were also present at either or both of two NASA-called consultant review meetings; the first was held on the afternoon of the Midterm Presentation on 1 March 1967, and the second took place in the General Dynamics offices in Washington, D.C., on 8 and 9 June 1967. Their assistance was very much appreciated by the Convair team.

Dr. Giacconi and Dr. Gursky of American Science and Engineering, Inc., assisted

Convair with the general scientific guidelines pertaining to the x-ray telescope. In addition, the following members of the scientific community, in the very beginning of the study, were helpful in the area of millimeter wavelength radio astronomy, magnetometers, and meteoroid collectors:

Dr. Allen H. Barrett,
Massachusetts Institute of Technology

Mr. W. Connor,
Jet Propulsion Lab.

Dr. Kenneth Hallam, GSFC

Dr. E. J. Smith,
Jet Propulsion Lab.

# TABLE OF CONTENTS

# SECTION 1

# INTRODUCTION

## 1.1 GENERAL

1.1.1 BACKGROUND. The purpose of this study was to identify and define three large space structure experiments through which the following flight objectives could be accomplished: evaluate the role of man in the deployment, assembly, alignment, maintenance and repair of large structures in space; evaluate the performance and behavior of large structures in space from a technology viewpoint; and provide a space structure which can be used to fulfill a "user oriented" requirement such as a radio astronomy antenna or solar cell array.

The logical point of departure for a study such as this is to first determine the most promising areas of science and technology which will probably require large structures in space. In viewing the potential NASA missions throughout the next decade, one can conclude that some of the more prominent requirements will evolve from the areas of astronomy, communications, and, to a lesser but still significant degree, from the requirements for solar cell arrays, micrometeoroid collectors, and magnetometers.

With regard to astronomy, regions of the electromagnetic spectrum which are of interest begin with the very long radio waves and continue through gamma rays. Although not all astronomers necessarily agree on which areas of the spectrum should receive the highest priority, general concensus on those bands of particular interest and specific recommendations for future astronomy in space are found in "Space Research Directions for the Future, Part 2" (Woods-Hole Report). This document and other appropriate sources of literature were used to guide this study in its relation to astronomy. A few of the more important conclusions and recommendations contained in the Woods-Hole Report are summarized in Figure 1-1. The darkened area

POTENTIAL APPLICATIONS OF LARGE STRUCTURES IN SPACE

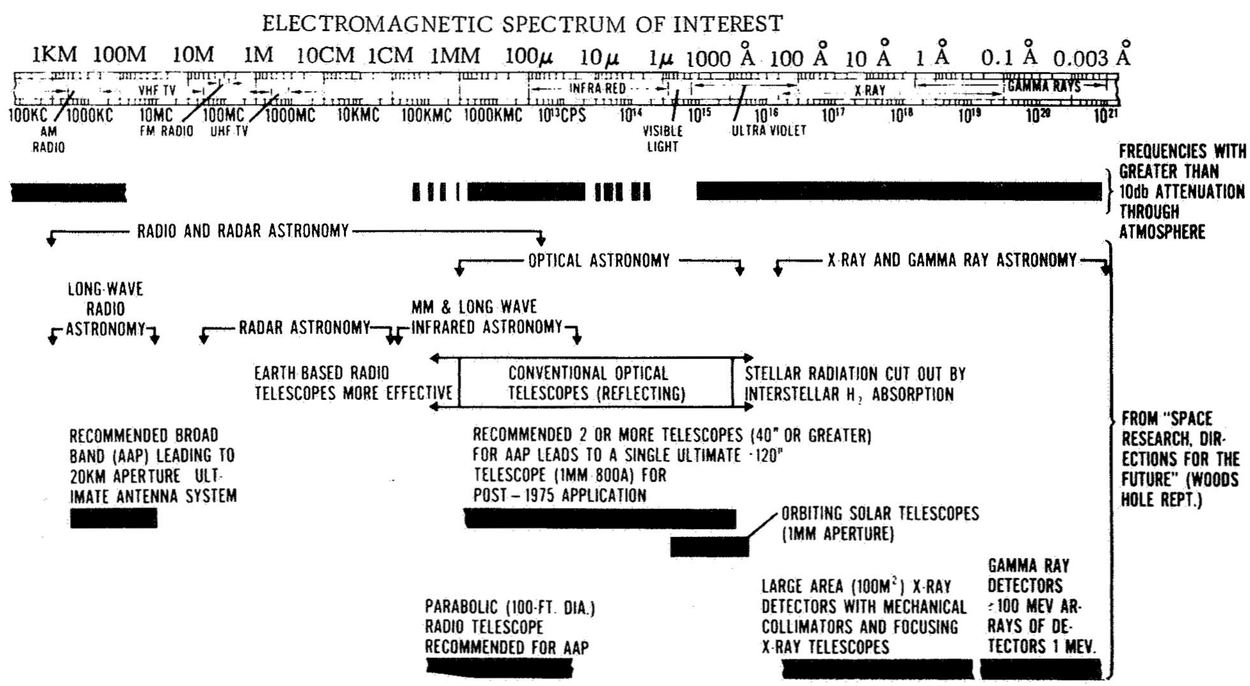



Figure 1-1

of the line at the top of the figure signifies those regions of the spectrum in which the atmospheric attenuation is greater than 10 db, and therefore those regions which are essentially blacked-out from the earth's surface. In these regions, astronomical observations are completely dependent on the ability to go into space. Accordingly, the Woods-Hole Report has made positive recommendations covering the entire spectrum with the exception of radar astronomy, which it says can be satisfied by ground-based observations.

Several of the bands of interest are particularly challenging to those interested in large space structures. First, consider the very long wave (10 M and longer) region. Antennas designed to operate in this region have two dominant characteristics: large physical dimensions and correspondingly large allowable tolerances. The Woods-Hole Report recommended a "Broad Band Antenna System leading to a 20 KM aperture ultimate antenna system." Although such an antenna would not be feasible during the AAP period, antenna types which may prove useful in the long wave region, and are therefore accounted for in this study, include log periodics, rhombics, broadside arrays, and phased arrays. In the submillimeter region the primary useful antenna concept is a fairly large parabolic radio telescope with very stringent tolerance requirements. A great deal of emphasis was therefore placed in this region of interest. Additional regions of interest to astronomers which involves large structures are the x-ray and gamma-ray parts of the spectrum. The Woods-Hole Report also contains specific large space structure requirements to support these astronomical programs.

Convair was directed by NASA to include in the study detailed analysis on four types of space structures relating to astronomy, long wave radio, submillimeter wave radio, and x- and gamma-ray astronomy. Although optical astronomy, including infrared, ultraviolet, and visible regions of the spectrum, is extremely important to future space flight, Convair was directed by NASA not to include these types of structures.

In the area of communications many varying types of potential candidate missions exist. They include TV broadcast, voice broadcast (ranging from direct to home broadcast to simple point to point relay), and deep space relay missions. Large space structure requirements vary widely with the mission and the potential time frame for application. Although work is continuing to develop high power, long life space power supplies such as radioisotopes and nuclear reactors, there will be a need for solar cell arrays, and so Convair was directed to include structures of this nature in the study. Additional areas which appeared to have the requirement for structures were magnetometer devices (the structural requirement emanates from the need to separate the magnetometer a long distance from the mother spacecraft) and micrometeoroid collectors.

In summary, it was directed by NASA that the concepts to be analyzed in the study be centered around those satisfying the following user-oriented applications requirements:

- Long Wave Radio Astronomy
- Millimeter Wave Radio Astronomy
- X- and $\gamma$-Ray Astronomy
- Communications
- Solar Cell Arrays
- Magnetometers
- Micrometeoroid Collectors

1.1.2 STUDY APPROACH. Figure 1-2 shows the major task areas to be accomplished during the study as directed by NASA during the contract orientation. The study was broken into two parts, the first half dealing with the analysis of a large number of candidate space structures concepts culminating in the selection of three which would then undergo more detailed analysis and design during the second half of the study.

1.1.3 SUMMARY OF RESULTS. Tasks 1 and 2 resulted in the preliminary design and analysis of 40 candidate space structure concepts for flight in the 1970-75 time frame. Three of these structures were selected at the midterm point of the study and were the subjects of detailed preliminary design and analysis during the second half of the study. They are: a long wave radio astronomy antenna called a Crossed-H Interferometer, a Focusing X-Ray Telescope, and a 100-foot Aperture Parabolic Antenna. An overall summary of the entire study is presented in this volume. Analysis of the 40 candidate structures can be found in Volume II, and detail analyses of the three selected concepts are contained in Volumes III, IV, and V, respectively.

## 1.2 INTRODUCTION TO VOLUME 1

This volume presents a summary of the entire study. Some of the primary guidelines were:

a. Concepts to be considered included the AAP 1968 to 1972 time period and the 1972 to 1975 time period. The 1968 to 1972 baseline astronaut requirements were assumed for 1972 to 1975 but requirements for improved capability were established.

b. Secondary emphasis was placed on solar cell arrays, magnetometers, and meteoroid collectors.

c. Experiments were defined for minimum cost. Although weight, volume, and packaging efficiency were considered, minimum cost was the prevalent consideration.

TASK AREAS

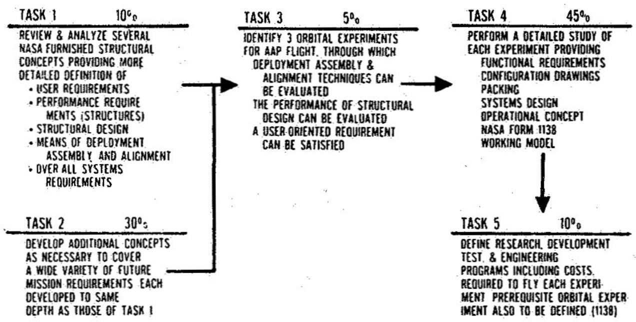



Figure 1-2

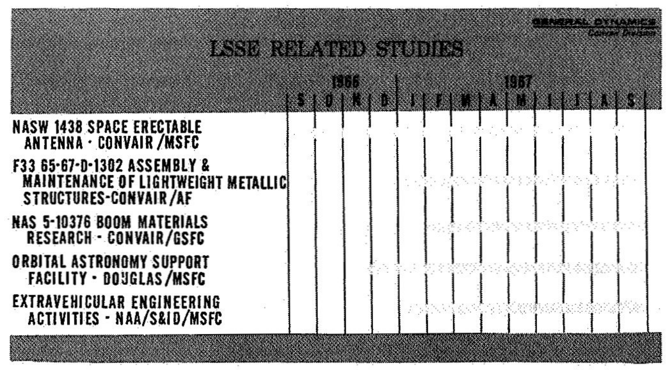



Figure 1-3

LSSE COORDINATION MEETINGS

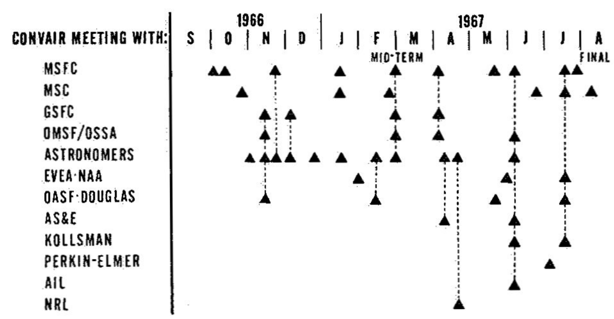



Figure 1-4

d. To the maximum extent feasible, the experiments were defined such that their performance was independent of any particular spacecraft configuration and mission profile.

To provide good data inputs to the study, extensive coordination with the scientific community, NASA, and industry was required, particularly for experiment user requirements and crew systems. See Figures 1-3 and 1-4.

1.2.1 MAN'S ROLE IN SUPPORT OF LARGE SPACE STRUCTURES. Baseline astronaut data and capabilities were established by the Crew Systems Division of MSC. Beneficial coordination meetings were held with the closely related NASA study of EVEA under Mr. Phil Pennington of NAA/S&ID.

A summary of man's recommended role is presented in Figure 1-5. In the event of electro/mechanical failure during deployment, he would be capable of stopping the deployment sequence and either replacing the failed component or completing the deployment manually. The study results indicate that man will be useful in alignment, maintenance, and repair, and essential to the task of refurbishment.

**ROLE OF MAN IN SUPPORT OF LARGE SPACE STRUCTURES (1970-75)**
**Antennas, X Ray & Optical Telescopes, Etc.**

- BACKUP DURING AUTOMATIC DEPLOYMENT
  - COMPLETE DEPLOYMENT MANUALLY
  - UNSCHEDULED REPAIR OF AUTOMATIC SYSTEM
- VISUAL INSPECTION BEFORE, DURING & AFTER DEPLOYMENT
- ALIGNMENT & CHECKOUT (INCLUDING CORRECTIVE ACTION)
- MAINTENANCE, REPAIR & REFURBISHMENT:
  - REPLACEABLE MODULE
  - REPLACEMENT OF STRUCTURAL MEMBERS
  - ADDITIVE (OVERLAY SYSTEMS)
  - RECHARGE SYSTEMS
- PERIODIC UPDATING OF INSTRUMENTATION/SENSORS (FOR EXPENSIVE, LONG-TERM ITEMS)
- REQUIRES PROPER DESIGN OF ITEM FROM HUMAN ENGINEERING VIEWPOINT
  - GROSS MANIPULATIONS
  - RESTRAINT
  - ACCESSIBILITY
  - SAFETY
  - HANDRAILS, ETC.
- PROPER TOOLS & OTHER SUPPORT EQUIPMENT
- SUPPORTING TESTS & SIMULATIONS

Figure 1-5

For man to enhance and update the performance of the systems and increase the probability of mission success, the design must consider human engineering from the beginning, particularly with respect to such considerations as EVA safety, restraints, accessibility, simplicity of task, and transfer of cargo.

1.2.2 SCIENTIFIC REQUIREMENTS AND CAPABILITY. To arrive at large space structures that fulfill the third main objective of the study, namely that of providing a scientifically useful device in earth orbit, an extensive investigation was undertaken to come up with scientific user requirements. These scientific requirements were then utilized both as evaluation criteria for already existing concepts and as design goals for generating the new concepts that were eventually selected by NASA for further detail design during the second half of the study. Several prominent members of the scientific community were instrumental in devising these design constraints. They are listed on page iv.

# SECTION 2

# ANALYSIS AND EVALUATION OF SPACE STRUCTURE CONCEPTS

## 2.1 LONG WAVE RADIO ASTRONOMY

2.1.1 USER REQUIREMENTS. Preliminary requirements used during the first half of the study are shown below.

a. Lifetime: Minimum of 1 year desired.

b. Orbit Altitude: Minimum of synchronous.

c. Effective Beamwidth: 100 deg$^2$ at 1 MHz desired - less than 10 deg in one direction, but could be greater for solar and planetary astronomy. Interferometers should be used, if possible, for improving this resolution to 2 deg.

d. Pointing Accuracy: One-half-beamwidth minimum to 1/10 for aspect determinations. In the case of a sweeping-mode or drift-mode antenna, the pointing direction must be known to within 1/10 half-power beamwidth or better.

e. Pointing Stability: Approximately 1/10 beamwidth or better.

f. Bandwidth: 500 kHz to 10 MHz desired, with emphasis on lower half.

g. Spectral Resolution: Good desired, and depends only on electronics for any one antenna.

h. Sensitivity: Unfilled apertures entirely adequate.

i. Lockon Time: One-half second to several hours for time-varying phenomena. For mapping observations, however, an antenna arrangement with as slow a drift rate as possible (up to approximately 1 deg/sec) suffices.

j. Tolerance: Prefer $1/20\tau$, but $1/16\tau$ is adequate (at 1 MHz, $\tau = 300$ m).

k. Orientation: Eliminate antenna pattern directional ambiguity.

2.1.2 ANALYSIS OF CONCEPTS. Eight concepts for long wave radio astronomy, Figure 2-1, were provided by NASA and industry. In addition, six new concepts, 51a through 54, were developed and analyzed to various levels of detail, depending on their predicted performance and feasibility. The predicted performance is provided in Table 2-1. The relative concept rating is a function of aperture area, beamwidths, bandwidths, and total structural deviation. A discussion of the evaluation process is provided in Section 2.8 and Volume II.

## 2.2 MM WAVE RADIO ASTRONOMY

2.2.1 USER REQUIREMENTS. Following is a summary of the requirements placed upon millimeter wavelength and infrared radio astronomy equipment:

a. Lifetime: 1 year minimum.

b. Orbit Altitude: Lower or high ok, with synchronous desirable.

c. Beamwidth: 1 sec of arc desired, but can be larger with aspect determinations.

d. Pointing Accuracy: 1 sec of arc.

e. Aspect Determination Interval: 1/10 of the beamwidth.

f. Jitter: $\pm 0.5$ beamwidth, when not using aspect determination.

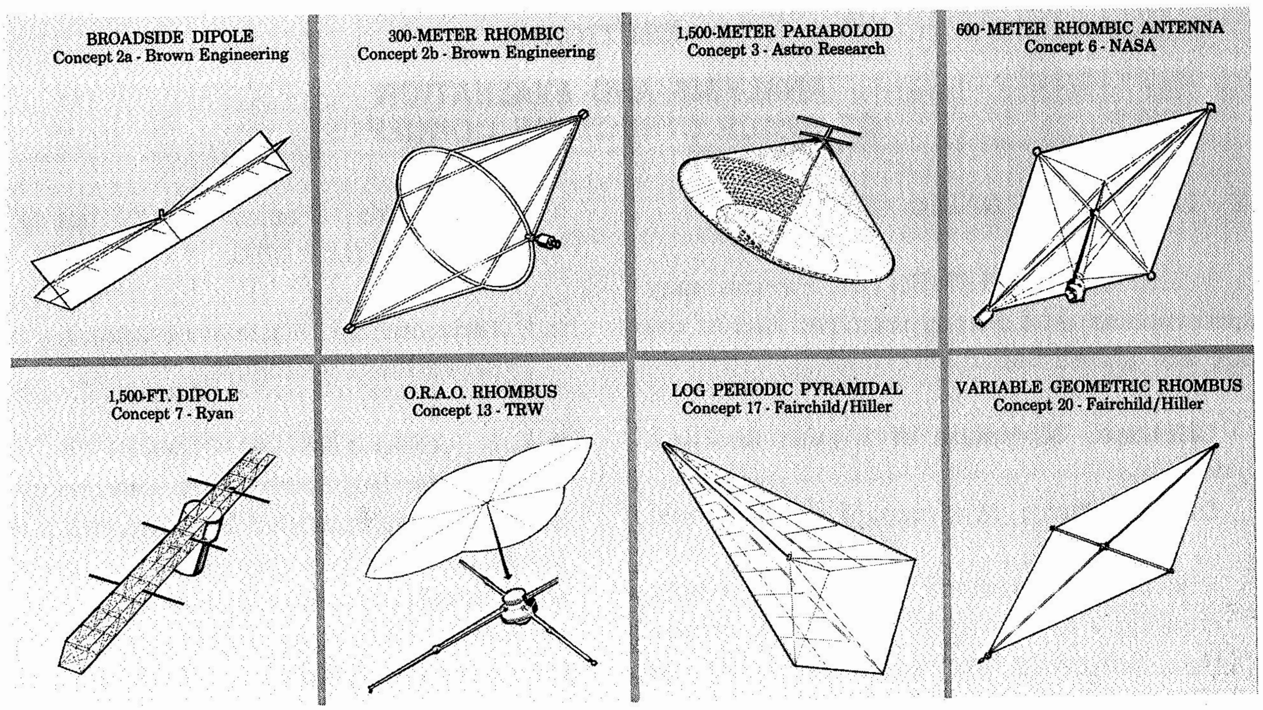



Figure 2-1

g. Sensor or Detector: Electronic receiver and/or super-cooled bolometer.

h. Bandwidth: Desired approximately 50 $\mu$ to 10 mm.

i. Spectral Resolution: 1° K desired.

j. Collecting Area: Parabolic dish, as large as possible.

k. Lockon Time: Several hours desired, but slow drift allowed for observation of stronger sources, using aspect determinations when beamwidth has considerably larger angular diameter than the source.

l. Surface Tolerance: Approximately 0.1 mil = 0.0025 mm for 50$\mu$ wavelength, and 2 mils for 1 mm wavelength diffraction limited capability.

m. Diffraction Limited: Down to 1 mm, at least. Preferred to 50$\mu$ = 0.05 mm. Diffraction limited to a certain wavelength entails keeping the surface tolerance of those parts of the antenna functioning at that wavelength to within 1/20 of the wavelengths.

n. Physical Dimensions: Parabolic dish of up to 100 ft diameter.

2.2.2 ANALYSIS OF CONCEPTS. For the time period under consideration there was no significant support from the scientific community for a mm-wave dish larger than 10 ft in diameter. Since this size, for the purpose of this study, was not considered a large structure, no structures were conceived for this application. However, some of the communications concepts were rated against mm-wave astronomy usage criteria.

## 2.3 X AND GAMMA RAY ASTRONOMY

2.3.1 USER REQUIREMENTS. The gamma-ray astronomy user requirements that may be feasibly fulfilled in the period of 1968-72 are:

a. Lifetime: Several months desired.

## Table 2-1. R-1, Long Wave Radio Astronomy

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL WEIGHT LBS | EXPERIMENT TOTAL VOLUME CU FT | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS: • (A) APERTURE AREA SQ FT | BEAMWIDTH • (F) DEG | BEAMWIDTH $f_1$ MHz | BEAMWIDTH DEG | BEAMWIDTH $f_2$ MHz | • (B) BANDWIDTH | • TOTAL STRUCTURE DEVIATION SURFACE | • TOTAL STRUCTURE DEVIATION (D) PERIM. DEFL. | MANEUVERABLE | • RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| R-1 | Long Wave Radio Astronomy | 24,000 | 4,500 | $2(10^4)$ | $3.2(10^7)$ | <10×10 | 0.5 | <<10×10 | 10 | 0.5-10 MHz | 4.8 FT | 48 FT | YES | | <2° For Interferometer |
| | | | | | | | | | | | | | | | |
| C-2a | Broadside Dipole Antenna | | | | | | | | | | | | | | Inflatable Version; Struct. & |
| | Array | 18,133 | 3,400 | $>2(10^4)$ | $1(10^6)$ | 26×90 | 1.0 | 9×47 | 3.0 | 1.0-3.0 MHz | | 43 FT | YES | | Mech. Feasible; 26×90 at 1 MHz |
| | Concept Rating | | | | 0.031 | 0.043 | | | | 0.210 | | 1.0 | | $2.8(10^{-4})$ | |
| | | | | | | | | | | | | | | | |
| C-2b | 300 Meter Rhombic Antenna | 8,370 | 1,890 | $>2(10^4)$ | $1.1(10^5)$ | 15×45 | 3.0 | 11×35 | 5.0 | 3-5 MHz | | 150 FT | YES | | 30°×30° at 4 MHz; 300M used |
| | Concept Rating | | | | 0.003 | 0.15 | | | | 0.210 | | 0.32 | | $3.03(10^{-5})$ | instead of 2400M; Requires |
| | | | | | | | | | | | | | | | 7000M for 10° at 1 MHz |
| | | | | | | | | | | | | | | | |
| C-3 | 1500 Meter Paraboloidal | 4,530 | 12,000 | $>2(10^4)$ | $5(10^5)$ | 68×68 | 0.5 | 2×2 | 10.0 | 0.5-10 MHz | | 24 FT | NO | | 3.5° at 4 MHz. If Reduced |
| | Antenna | | | | | | | | | | | | | | to 400M Dia, Vol = 5720, Beam |
| | Concept Rating | | | | 0.0156 | $2.2(10^{-2})$ | | | | 1.00 | | 1.0 | | $3.4(10^{-4})$ | 30° at 2.3 MHz |
| | | | | | | | | | | | | | | | |
| C-6 | 600 Meter Rhombic Antenna | 15,600 | 2,000 | $>2(10^4)$ | $6.3(10^5)$ | 30×90 | 0.5 | 5×15 | 10.0 | 0.5-10 MHz | | 10 FT | YES | | *Does not include Att. Control |
| | Concept Rating | | | | 0.020 | 0.037 | | | | 1.0 | | 1.0 | | $7.4(10^{-4})$ | Prop; 30°×30° at 2 MHz |
| | | | | | | | | | | | | | | | |
| C-7 | Member Structure for Large | | | | | | | | | | | | | | |
| | Astronomy Antenna | 2,600 | 2,300 | $>2(10^4)$ | $7.8(10^4)$ | 20×90 | 1.23 | 7×40 | 5.35 | 1.23-5.35 MHz | | 30 FT | YES | | |
| | Concept Rating | | | | $2.4(10^{-3})$ | $5.5(10^{-2})$ | | - | | 0.434 | | 1.0 | | $5.7(10^{-5})$ | 13°×78° at 3 MHz |
| | | | | | | | | | | | | | | | |
| C-13 | Orbiting Radio Astronomical | | | | | | | | | | | | | | |
| | Observatory | 400 | 30 | 500 to 32,500 | $1.2(10^7)$ | 9×28 | 0.5 | 5×18 | 10 | 0.225-10 | | 9.6 | NO | | |
| | Concept Rating | | | | 0.38 | $4(10^{-1})$ | | | | 1.0 | | 1.0 | | $1.4(10^{-1})$ | |
| | | | | | | | | | | | | | | | |
| C-17 | Log Periodic Pyramidal | (2,084) | 100 | $>2(10^4)$ | $3(10^6)$ | 50×60 | 0.5 | 50×60 | 10 | 0.5-10 MHz | | 30 | YES | | *10° Resolution in Interfer- |
| | Concept Rating | | | | 0.094 | 0.033 | | | | 1.0 | | 1.0 | | $3.1(10^{-3})$ | ometer Configuration |
| | | | | | | | | | | | | | | | |
| C-20 | Extensible Rhombic | 18,000 | 1000 | $2(10^4)$ | $1.7(10^6)$ | 30×90 | 0.5 | 7×21 | 10 | 0.5-10 MHz | | (200) | YES | $4.2(10^{-3})$ | |
| | Concept Rating | | | | 0.053 | 0.33 | | | | 1.0 | | 0.24 | | | |
| | | | | | | | | | | | | | | | |
| C-51a | Colinear Dipole Array, 5 Km | 2,084 | 3,500 | $2(10^4)$ | $7.49(10^7)$ | 13×360 | 0.25 | 3×360 | 1.25 | 0.25-1.25 MHz | | 45 FT | | | |
| | Concept Rating | | | | 1.0 | $2.14(10^{-2})$ | | | | $7.9(10^{-2})$ | | 1.0 | | $1.7(10^{-3})$ | |
| | | | | | | | | | | | | | | | |
| C-51b | Crossed-H | | | | | | | | | | | | | | Interferometer Lobes 3.3° at |
| | Interferometer | 3,657 | 1280 | $2(10^4)$ | $1.07(10^6)$ | 90×225** | 0.5 | 40×80** | 10 | 0.5-10 MHz | | 10 FT | | | 0.5 MHz and 0.7° at 4 MHz |
| | Concept Rating | | | | $3.34(10^{-2})$ | $5(10^{-3})$ | | | | 1.0 | | 1.0 | | $1.7(10^{-4})$ | |
| | | | | | | | | | | | | | | | |
| C-52a | 1500 Meter Rhombic Antenna | | | | | | | | | | | | | | |
| | Ext Struct | 23,866 | 4,100 | $2(10^4)$ | $5.45(10^6)$ | 17×50 | 0.5 | 5×15 | 10 | 0.5-10 MHz | | 50 FT | | | |
| | Concept Rating | | | | $1.70(10^{-1})$ | 0.118 | | | | 1.0 | | 0.96 | | $1.9(10^{-2})$ | |
| | | | | | | | | | | | | | | | |
| C-52b | 1200 Meter Rhombic Antenna, | | | | | | | | | | | | | | |
| | Int Struct | (22,000) | 4,100 | $2(10^4)$ | $3.84(10^6)$ | 20×55 | 0.5 | 5×15 | 10 | 0.5-10 MHz | | 50 FT | | | |
| | Concept Rating | | | | $1.2(10^{-1})$ | $9.1(10^{-2})$ | | | | 1.0 | | 0.96 | | $1.1(10^{-2})$ | |
| | | | | | | | | | | | | | | | |
| C-53 | End Fire Antenna 3 Km Array | 4,000 | 3,800 | $2(10^4)$ | $4.5(10^7)$ | 30×30 | 0.25 | 14×14 | 1.25 | 0.25-1.25 MHz | | | | | Rejected for Excessive EVA |
| | Concept Rating | | | | 1.0 | 0.11 | | | | 0.105 | | | | | and Poor Dynamics. Work |
| | | | | | | | | | | | | | | | Discontinued on Concept |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

** PLUS INTERFEROMETER RESOLUTION TO 1.7° AT 0.5 MHz AND 0.35° AT 2.5 MHz AND ABOVE

$$\text{CONCEPT RATING} = \frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - (\ )}{C - (\ )}$$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

b. Orbit Altitude: 200 n. mi. or lower, or well above radiation belts.

c. Field of View: Two degrees desired, but larger field is acceptable, depending on synthesis techniques.

d. Resolution: Accomplished by aspect determination or occultation.

e. Pointing Accuracy: <1° or ±1/2 field of view or better. Alternative target and bandground observations desirable.

f. Aspect Determination Interval: 1/10 field of view or better.

g. Sensor or Detector: Anticoincidence shielded scintillation counters, spark chambers, or Cerenkov counters.

h. Bandwidth: 40 kev - 20 Mev using anticoincidence shielded scintillation counters; 30 Mev - 300 or 400 Mev using spark chambers. Above 0.5 Bev gas-filled Cerenkov counters.

i. Spectral Resolution: 10 per cent of frequency.

j. Sensitivity: $10^{-3}$ photon/cm$^2$-sec-ster to $10^{-8}$ photon/cm$^2$-sec-ster.

k. Effective Collecting Area: Up to 10 M$^2$, but 1 M$^2$ suffices.

l. Lockon Time: 6 - 144 hr; sweep with aspect determination may be employed when possible.

m. Structural Tolerance: The entire array must be mechanically aligned to within the capability provided by aspect pointing knowledge.

The user requirements for x-ray astronomy proportional counter arrays are:

a. Lifetime: One year.

b. Orbit Altitude: 200 n. mi. or lower preferred. 0° - 28.5° inclination.

c. Field of View: One deg maximum.

d. Resolution: Can be improved to arc seconds by use of ODA type collimator, sweeping across the source, and synthesizing the direction of the incoming radiation. By occultation to about 1' arc.

e. Pointing Accuracy: $< \pm 0.50°$; prefer $< \pm 6'$ arc. Alternative target and background looks.

f. Aspect Determination: Required as fine as possible; at least 0.5' arc in scan mode.

g. Sensor or Detector: Gas-filled proportional counters, anticoincidence arrangement required.

h. Bandwidth: 1 - 20 kev desired.

i. Spectral Resolution: 40 per cent at 1 kev to 20 per cent at 10 kev desired.

j. Sensitivity: 1 - 10 kev range demands minimum power detectable of about $10^{-2}$ photons/cm$^2$-sec-ster. 10 - 40 kev range demands about $10^{-4}$ photons/cm$^2$-sec-ster.

k. Effective Collecting Area: As large as possible, up to 100 m$^2$.

l. Lockon Time: 5 - 360 minutes, but drift modes may be employed up to 1°/sec.

m. Structural Tolerance: The field of view accomplished by collimators. However, the entire array must be mechanically aligned to within the resolution capability provided by mechanical collimators.

n. Physical Dimensions: Individual cell elements 2 ft by 2 ft by 8 in. deep, plus electronics and structure

The user requirements for large imaging grazing incidence x-ray telescopes are:

a. Lifetime: At least a year.

b. Orbit: 260 n. mi. or lower circular, 28.5° inclination.

c. Field of View or Beamwidth: About 10' to 30' arc.

d. Angular Resolution: 2" arc or better; depends on jitter rate and number of exposures per second.

e. Pointing Accuracy: 5' arc offset.

f. Lockon Capability; Within a 1' arc square.

g. Jitter: The jitter must not exceed the angular resolution desired.

h. Sensors or Detectors: Image intensifier plus film or electronic imaging tube. Spectrometer. Polarimeter.

i. Bandwidth: Approximately 2 Å to 300 Å.

j. Spectral Resolution: $\tau/\Delta\tau$ = 100 to 1000 or better.

k. Collecting Area: 200 $cm^2$ minimum effective collecting area.

l. Lockon Time: 10 minutes to several hours.

m. Data: 5 to 10 x 106 bits per orbit.

n. Structural Tolerance: Paraboloidal-hyperboloidal reflecting surfaces must by precision machined and the electro-deposited nickel or flame-sprayed nickel oxide optically polished to reflect grazing incidence x-rays down to at least 2 Å wavelength. The alignment of the reflecting mirrors with the focal point must be kept so that the image remains entirely on the detector sensing area.

o. Physical, Unstowed, Dimensions: Up to 10 ft diameter by up to 100 ft long.

2.3.2 ANALYSIS OF CONCEPTS. Structures for x-ray and gamma-ray detectors were conceived and analyzed. Major characteristics are compared in Table 2-2. Concept 71, shown in Figure 2-2, consists of 352 square feet of x-ray collimators mounted in a single plane and gamma ray detectors with the same duration.

## 2.4 COMMUNICATIONS

2.4.1 USER REQUIREMENTS. User requirements for communications systems are provided in Table 2-3.

Table 2-2. Concept Comparisons

| CONCEPT | GAMMA RAY MODULES (sq ft) | X-RAY MODULES (sq ft) | X-RAY TELESCOPE (ft Dia) | WEIGHT (lb) | ATTITUDE CONTROL SYSTEM | STOWAGE |
|---|---|---|---|---|---|---|
| 71 | 11 | 352 | — | 13,118 | LEM - Moment Wheels | Below Rack |
| 72 | — | 1000 | — | 16,576 | CSM - Moment Sheels | Below and Above Rack |
| 73 | — | — | 10 | 22,000 | Own | Entire SLA |

X-RAY AND GAMMA-RAY CONCEPT 71

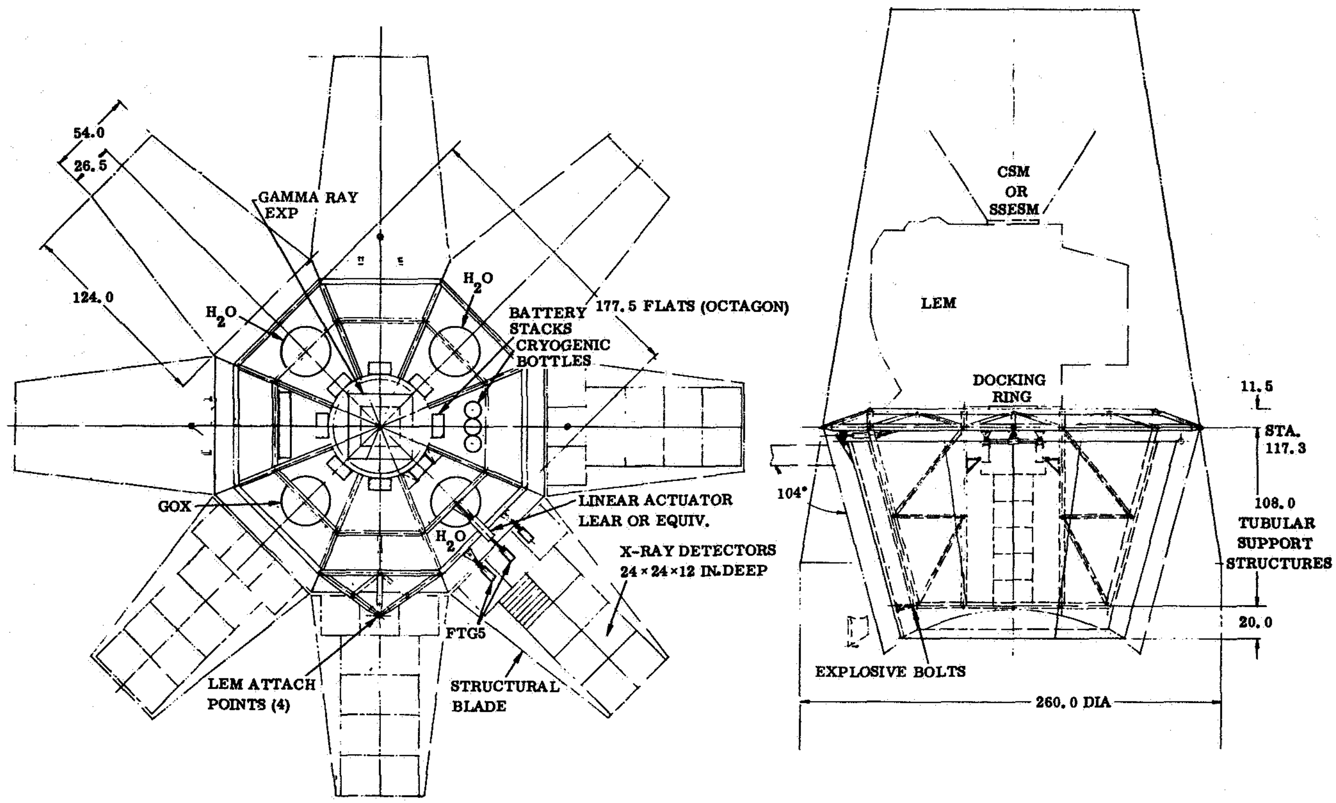



Figure 2-2

2.4.2 ANALYSIS OF CONCEPTS. Eight concepts for communications, provided by NASA and industry, were analyzed. Nine antenna concepts from related Convair studies were added for evaluation. Most of these concepts are depicted in Figure 2-3. A comparison of predicted performance and rating when measured against one specific application, TV broadcast, is provided in Table 2-4.

## 2.5 SOLAR CELL ARRAYS

2.5.1 USER REQUIREMENTS. Electrical power is required for scientific, electronics, and spacecraft systems including data and communications equipment. Power requirements for communication satellites may vary from a few watts to a megawatt or more. Postulated power requirements for various applications after 1972 are depicted in Figure 2-4.

2.5.2 ANALYSIS OF CONCEPTS. Solar cell arrays were provided by Ryan, Fairchild/Hiller, and a NASA funded study by Boeing. Additional arrays were conceived for application to the Concept 75 communications antenna and the S-IVB Workshop. Concepts and performance summary are shown in Figure 2-5 and Table 2-5.

## 2.6 MAGNETOMETERS

2.6.1 MAGNETOMETER USER REQUIREMENTS. Magnetometers must be isolated from large structures and mounted on non-ferromagnetic supports in such a position that disturbances in the surrounding plasma due to the spacecraft do not interfere with the measurements. Usually the magnetometer should be separated from the spacecraft by about three times the largest spacecraft dimension to escape large loops of current.

Table 2-3. Communication System Space Antenna/Transmitter Requirements

| FREQUENCY (Hz) | BEAMWIDTH | GAIN (db) | SIZE (ft) | BANDWIDTH (Hz/Channel) | TRANSMITTER POWER OUTPUT (watt) | ALTITUDE | LIFETIME | REMARKS |
|---|---|---|---|---|---|---|---|---|
| 1. Space Communications to Provide Instant Contact with most of Major Nations | | | | | | | | |
| 1G-10G | 17° | 19 | 0.4 - 4 | 4000 | 100 | Synch | 2-10 yr | 125 Duplex Channels |
| 2. Voice Broadcast Direct to Home | | | | | | | | |
| 15M-30M | 30° | 15 | 104x104x15 | 10k | 5k | Medium | 2-10 yr | |
| 88M-108M | 17° | 19 | 40 | 10k-150k | 5k | Synch | 2-10 yr | 5 Channels |
| 88M-108M | 20° | 18 | 35 | 19k-150k | 1k | Medium | 2-10 yr | 5 Channels |
| 3. Real-Time TV Direct to Home (20 db Ground Antenna, 6 db NF Preamp) | | | | | | | | |
| 900 M (Approx.) | 3.5° x 7.4° | 30 | 22 x 11 | 6M | 57k | Synch | 2-10 yr | U. S. Coverage 3 Channels |
| 900M (Approx.) | 2.6° x 3.5° | 35 | 29 x 22 | 6M | 20k | Synch | 2-10 yr | 1-hr time zone in U. S. (3 Channels) |
| 900M (Approx.) | 0.5° | 50 | 150 | 6M | 1k | Synch | 2-10 yr | 200 x 200 Mi.$^2$ (5 Channels) |
| 4. Real-Time TV to Distribution Center (42 db Ground Antenna, 3 db NF Preamp) | | | | | | | | |
| 0.9G to 4G | 3.5° x 7.4° | 30 | 22 x 11 | 6M | 600 | Synch | 5-10 yr | U. S. Coverage (10 Channels) |
| 0.9G to 4G | 2.6° x 3.5° | 35 | 29 x 22 | 6M | 200 | Synch | 5-10 yr | 1-hr time zone in U. S. (10 Channels) |
| 0.9G to 4G | 0.5° | 50 | 150 | 6M | 6 | Synch | 5-10 yr | 200 x 200 Mi.$^2$ (10 Channels) |
| 5. Telephone Communications, Personal Nature, Global | | | | | | | | |
| 1G-4G | 17° | 19 | 1-4 | 4K | 1.25K | Synch | 10 yr | Hemispherical Coverage (1000 Channels) |
| 1G-4G | 1° | 44 | 20-70 | 4K | 500 | Synch | 10 yr | Multiple Beam Coverage (1000 Channels) |
| 6. Spacecraft to Spacecraft Communications | | | | | | | | |
| 12G-100G | 0.5° | 50 | 20 | | 10K | 200-240,000 mi. | | |
| 7. Orbiting Relays for Communications from Deep Space Probes | | | | | | | | |
| 2G-100G | 0.1° | 64 | 170-20 | $10^7$ bps | 10K | 200-22,300 mi. | 3 yr | |
| 8. High Data Rate, Point-to-Point Communications | | | | | | | | |
| 2G-10G | 0.2° | 58 | 170-35 | $10^6$ bps | 10K | 22,300 | 3 yr | |
| 9. Location Determination | | | | | | | | |
| 7G-8G | 0.25° | 56 | 40 | | 0.5K | 22,300 | 2 yr | |

The magnetometer support boom must be compatible with the characteristics of the magnetometer user requirements, which are:

a. Lifetime: Commensurate with that of spacecraft.

b. Orbit Altitude: Within the magnetosphere.

c. Orbit Inclination: Perpendicular and parallel to the earth-sun line.

d. Orbit Eccentricity: High for magnetic lapse rate measurements, low for magnetic isodistance lines.

e. Field of Influence: Magnetosphere first priority.

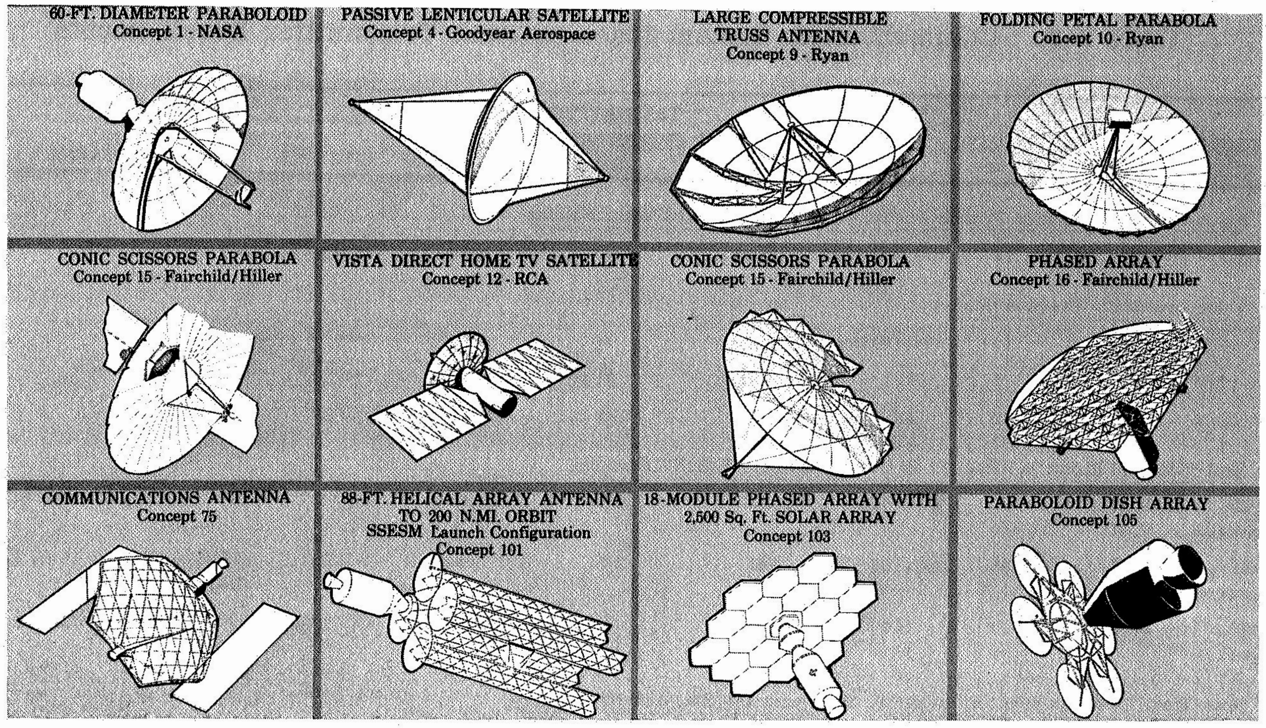



Figure 2-3

f. Pointing Accuracy: Approximately 1° at high altitudes, 3° - 5° at low altitudes.

g. Sensor or Detector: Rubidium-vapor or flux gate magnetometer.

h. Field Strength: 20, 000 - 100, 000 gamma at lower altitudes to approximately 100 gamma at high altitudes.

i. Sensitivity: <1000 gamma for low altitudes to 1 - 5 gamma variations at high altitudes.

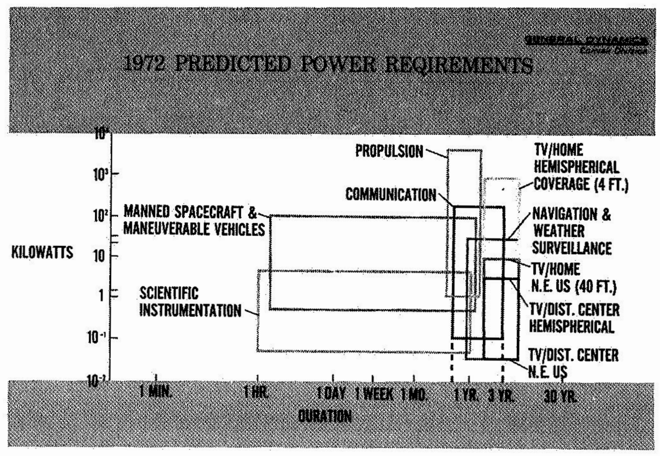



Figure 2-4

j. Flux Directions Measured: $4\pi$ steradians.

k. Size: Approximately 1 ft diameter by 6 in. high.

l. Structure Tolerance: Only in instrument itself.

m. Deployment: Extend on long boom, about three times as long as the largest spacecraft dimension, to avoid magnetic materials.

n. Operation: Continuous.

o. Important Measurements: Solar flare magnetosphere modulations, lunar and terrestrial magnetic tail or wake studies.

p. Data Rate: 100 bps for low altitudes - scalar magnetometer. 1000 bps for high altitudes - vector magnetometer.

q. Power Requirement: Approximately 5 w for each scalar axis.

# Table 2-4. R-7 TV Broadcast

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL: WEIGHT LBS | EXPERIMENT TOTAL: VOLUME CU FT | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS: * (A) APERTURE AREA SQ FT | BEAMWIDTH: $f_1$ MHz | BEAMWIDTH: * (F) DEG | BEAMWIDTH: $f_2$ MHz | BEAMWIDTH: DEG | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION: SURFACE | * TOTAL STRUCTURE DEVIATION: (D) PERIM. DEFL. | MANEUVERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| R-7 | TV Broadcast | 24,000 | * 4,500 | $2(10^4)$ | $6(10^3)$ | $8(10^2)$ | 3.5×7.4 | $9(10^2)$ | 3.5×7.4 | 800-900 MHz | 0.6 IN. | 6 IN. | YES | | Multiple beams, one per |
| | | | | | | | | | | | | | | | broadcast area |
| C-1 | 60 Ft Paraboloid Antenna | 9,100 | 1,980 | $2(10^4)$ | $1.4(10^3)$ | $8(10^2)$ | 1.4×1.4 | $9(10^2)$ | 1.3×1.3 | 800-900 MHz | 0.03 IN. | 0.3 IN. | YES | | |
| | Concept Rating | | | | 0.233 | | 1.0 | | | 1.0 | | 1.0 | | $2.33(10^{-1})$ | |
| C-4 | Lenticular Communications | | | | | | | | | | | | | | |
| | Satellite | 1,484 | 78 | $2(10^4)$ | ← | | NOT APPLICABLE** | | | → | | | | | |
| | Concept Rating | | | | | | | | | | | | | | |
| C-9 | Large Compressible Member | | | | | | | | | | | | | | |
| | Truss Struct for Space | | | | | | 0.86× | | 0.77× | | | | | | Electrical power supply |
| | Antenna | 6,527 | 1,800 | $2(10^4)$ | 4,250 | $8(10^2)$ | 0.86 | 900 | 0.77 | 800-900 MHz | | 0.2 IN. | YES | | can vary from 500 to 2,200 lb |
| | Concept Rating | | | | 0.708 | | 1.0 | | | 1.0 | | 1.0 | | $7.08(10^{-1})$ | |
| C-10 | Folding Petal Struct for | | | | | | | | 1.55× | | | | | | Electrical power supply |
| | Large Space Antenna | 3,015 | 1,000 | $2(10^4)$ | 1000 | 800 | 1.7×1.7 | 900 | 1.55 | 800-900 MHz | 0.1 IN. | 0.1 IN. | YES | | can vary from 500 to 2,200 lb |
| | Concept Rating | | | | 0.167 | | 1.0 | | | 1.0 | | 0.1 | | $1.67(10^{-1})$ | |
| C-11 | 30 Ft Petal Paraboloid | | | | | | | | | | | | | | Electrical power supply can |
| | Antenna | 3,018 | 1,200 | $2(10^4)$ | 400 | 800 | 2.9×2.9 | 900 | 2.6×2.6 | 800-900 MHz | 0.05 IN. | 0.2 IN. | | | vary from 500 to 2,200 lb |
| | Concept Rating | | | | 0.066 | | 1.0 | | | 1.0 | | 1.0 | | $6.6(10^{-2})$ | |
| C-12 | Vista Direct Home TV | | | | | | | | | | | | | | |
| | Broadcast | 4,255 | 14,730 | $2(10^4)$ | 1,000 | 800 | 1.7×1.7 | 810 | 1.7×1.7 | 800-810 MHz | | 0.33 IN. | YES | | |
| | Concept Rating | | | | 0.167 | | 1.0 | | | 0.1 | | 1.0 | | $1.67(10^{-2})$ | |
| C-15 | Conic Scissors Parabola | | | | | | | | | | | | | | Electrical power supply can |
| | Array | < 2,000 | 1,000 | $2(10^4)$ | 400 | 800 | 2.9×2.9 | 900 | 2.6×2.6 | 800-900 MHz | 0.03 IN. | 0.2 IN. | YES | | vary from 500 to 2,200 lb, |
| | Concept Rating | | | | 0.066 | | 1.0 | | | 1.0 | | 1.0 | | $6.6(10^{-2})$ | |
| C-16 | Bi-Directional Scissors Array | 1,250 | 200 | $2(10^4)$ | 2,930 | 90 | 10×10 | 150 | 6×6 | 90-150 MHz | | 1.5 IN. | YES | | Lacks Sizing |
| | Concept Rating | | | | 0.49 | | 0.259 | | | 0 | | 1.0 | | 0 | Reject: Frequency does not lie |
| | | | | | | | | | | | | | | | within required band |
| C-75 | Commercial TV, 150 Ft Dia | | | | | | | | | | | | | | |
| | (Fager) | 7,629 | 3,040 | $2(10^4)$ | 9,500 | 800 | 0.58×0.58 | 900 | 0.51×0.51 | 800-900 MHz | | 0.3 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 1.0 | | | 1.0 | | 1.0 | | 1.0 | |
| C-100a | 50 Ft Trussed Parabolic | | | | | | | | | | | | | | |
| | Antenna with 750 $Ft^2$ Solar | | | | | | | | | | | | | | |
| | Array | 2,000 | 570 | $2(10^4)$ | 1,000 | 800 | 1.7×1.7 | 900 | 1.55×1.55 | 800-900 MHz | | 0.055 IN. | YES | | Beryllium Struct |
| | Concept Rating | | | | 0.167 | | 1.0 | | | 1.0 | | 1.0 | | $1.67(10^{-1})$ | |
| C-100b | 50 Ft Trussed Parabolic | | | | | | | | | | | | | | |
| | Antenna with 2,400 $Ft^2$ Solar | | | | | | | | | | | | | | |
| | Array | 2,600 | 500 | $2(10^4)$ | 1,000 | 800 | 1.7×1.7 | 900 | 1.55×1.55 | 800-900 MHz | | 0.055 IN. | YES | | Beryllium struct |
| | Concept Rating | | | | 0.167 | | 1.0 | | | 1.0 | | 1.0 | | $1.67(10^{-1})$ | |
| C-100c | 50 Ft Trussed Parabolic With | | | | | | | | | | | | | | Beryllium struct |
| | Directed Solar Array | 1,615 | 570 | $2(10^4)$ | 1,000 | 800 | 1.7×1.7 | 900 | 1.55×1.55 | 800-900 MHz | | 0.055 IN. | YES | | |
| | Concept Rating | | | | 0.167 | | 1.0 | | | 1.0 | | 1.0 | | $1.67(10^{-1})$ | |
| C-101 | 78 Ft Contrawound Helix | | | | | | | | | | | | | | |
| | (Fager) | 7,870 | 1,440 | $2(10^4)$ | $1.1(10^4)$ | 15 | 32×32 | 75 | 14.5×14.5 | 15-75 MHz | | 0.22 IN. | YES | | Reject: Frequency does not lie |
| | Concept Rating | | | | 1.0 | | $2.53(10^{-2})$ | | | 0 | | 1.0 | | 0 | within required band |
| C-102 | 100 Ft Dia Trussed Para- | | | | | | | | | | | | | | |
| | bolic Antenna With 2,100 $Ft^2$ | | | | | | | | | | | | | | |
| | Fixed Cell Solar Array | 7,570 | 5,450 | $2(10^4)$ | 4,000 | 800 | 0.86×0.86 | 900 | 0.77×0.77 | 800-900 MHz | | 0.1 IN. | YES | | |
| | Concept Rating | | | | 0.666 | | 1.0 | | | 1.0 | | 1.0 | | $6.66(10^{-1})$ | |
| C-103 | Horn Phased Array With | | | | | | | | | | | | | | |
| | 2,500 $Ft^2$ Fixed Cell Solar | | | | | | | | | | | | | | |
| | Array | 14,800 | 5,450 | $2(10^4)$ | 2,410 | 800 | 1.5×1.5 | 900 | 1.5×1.5 | 800-900 MHz | | 0.19 IN. | YES | | |
| | Concept Rating | | | | 0.402 | | 1.0 | | | 1.0 | | 1.0 | | $4.02(10^{-1})$ | |
| C-104 | 7-Petalated 30 Ft Dia | | | | | | | | | | | | | | |
| | Antennas With RTG Power | 12,625 | 5,450 | $2(10^4)$ | 2,240 | 800 | 1.2×1.2 | 900 | 1.0×1.0 | 800-900 MHz | | 0.125 IN. | YES | | |
| | Concept Rating | | | | 0.373 | | 1.0 | | | 1.0 | | 1.0 | | $3.73(10^{-1})$ | |
| C-105 | Seven Dish Paraboloid TV | | | | | | | | | | | | | | |
| | Broadcast Antenna With Solar | | | | | | | | | | | | | | |
| | Array | 1,236 | 4,500 | $2(10^4)$ | 850 | 800 | 1.9×1.9 | 900 | 1.7×1.7 | 800-900 MHz | 0.006 IN. | 0.116 IN. | YES | | |
| | Concept Rating | | | | 0.142 | | 1.0 | | | 1.0 | | 1.0 | | $1.4(10^{-1})$ | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

$$\text{CONCEPT RATING} = \frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - (\,)}{C - (\,)}$$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

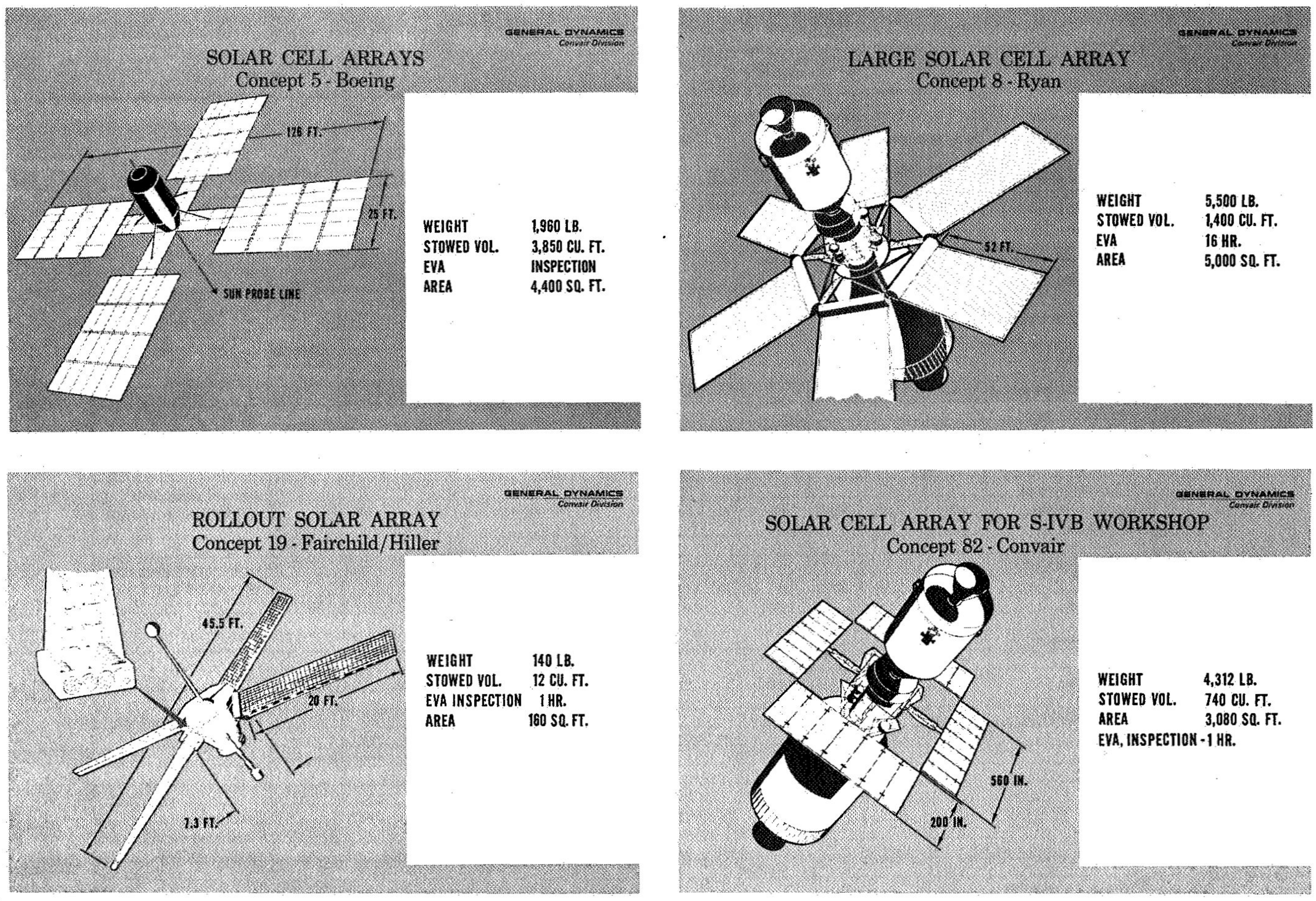



Figure 2-5

Table 2-5. Summary of Solar Array Concept Performance

| CONCEPT NO. | 5 | 8 | 19 | 81 | 82 |
|---|---|---|---|---|---|
| FACTOR | BOEING | RYAN | FAIRCHILD/ HILLER | | |
| Gross Area, $ft^2$ | 5000 | 5000 | 160 | 3000 | 3080 |
| Application | Mars Mission | (S-IVB) | — | TV Broad- cast/Comm Antenna | S-IVB |
| Net Area, $ft^2$ | 4433 | 3500 | 147 | 2700 | 2770 |
| Gross Generated Power, kw | 47.7 | 15.8 | 1.7 | 30 | 27 |
| Specific Output, watts/$ft^2$ | 10.8 | 4.5 | 10 | 11.1 | 10.1 |
| watts/lb | 24.3 | 5.2 | 12.1 | 9.3 | 6.3 |
| *Gross Weight | 1960 | 3060 | 140 | 3238 | 4312 |
| Stowed Volume, $ft^3$ | 3832 | 1400 | 150 | 1800 | 1400 |
| Deployed Dimension, ft | 25 × 28 (4 Panels) | 16 × 53 (6 Panels) | 2 × 20 (4 Panels) | 28 × 54 (2 Panels) | 17 × 47 (4 Panels) |

*Does not include batteries, conditioning equipment, or cabling.

r. Thermal Constraint: Preamplifier heaters required, or preamplifier redesign necessary to withstand -54°C to +65°C.

2.6.2 ANALYSIS OF CONCEPT 91, MAGNETOMETER BOOM. Only one concept was analyzed. This concept consists of a magnetometer boom attached to a petal of a Concept 72 spacecraft. The boom is composed of 11 telescoping sections that extend 150 ft from the perimeter of the spacecraft. Each of the sections is triangular in cross section (see Figure 2-6).

Attitude control of the end of the boom consists of keeping it aligned with the supporting spacecraft. Attitude control is provided to the extent required by boom deflections by vernier servo adjustments on two axes. In its launch configuration the boom is stowed in a package approximately 12 in. by 14 in. by 200 in., and weighs approximately 200 lb.

2.7 METEOROID COLLECTORS

2.7.1 USER REQUIREMENTS. Three basic types of meteoroid detectors are of interest: gelatin, wire mesh, and crystal. These detectors permit determination of mass and velocity, allow crater characteristics to be studied, and enable recovery of undamaged particles from emulsions.

User requirement data are:

a. Lifetime: 1 year.

b. Orbits: Altitude: various, depending on investigation desired. Inclination: normal to earth's trajectory. Eccentricity: preferably circular (zero).

c. Field of View: $2\pi$ steradians.

d. Directional Resolution: Desired to ±10 deg, but not necessary.

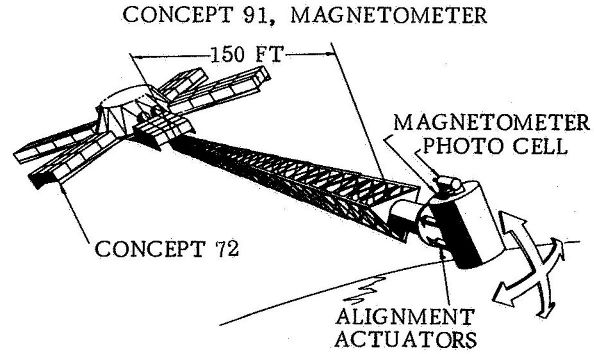



Figure 2-6

e. Pointing Accuracy: Within 10-deg array should point perpendicular to earth's trajectory.

f. Detector: Gelatin emulsions, thin plates, membranes, wire grids, or crystals.

g. Sensitivity: Depends on individual investigation.

h. Collecting Area: As large as possible (up to 15,000 sq ft for some investigations).

i. Physical Dimensions: In case of wire grids and emulsions, individual cells of 2 ft by 2 ft.

j. Structural Tolerance: To synthesize the incoming directions of several particles, individual cells must be aligned within resolution capability.

k. Significant Measurements: Meteor stream peaks; fuel tanks and wall panel penetration.

l. EVA: Post-micrometeoroid detector replacement, handling, and maintenance; removal and replacement of particle collectors possibly weekly or monthly.

m. Materials: For some applications, gelatin, which may have to be kept warm to remain gelatinous.

n. Electrical Power: 10 watts/sq meter.

2.7.2 ANALYSIS OF CONCEPTS. Two meteoroid detector concepts were provided by industry, and one meteoroid collector was conceived for analysis and evaluation. These are shown in Figure 2-7. Detailed descriptions and analysis are provided in Volume II.

## 2.8 EVALUATION OF CONCEPTS

This section presents a brief summary of the results of the evaluation of over 40 large space structures concepts. Evaluation criteria were developed in close coordination with the NASA/MSFC project monitor. They included performance, weight, volume, cost, schedule, feasibility, and technology contribution.

2.8.1 USER REQUIREMENT CONCEPT RATING. Performance predicted for long wave radio astronomy and communications concepts were provided in Tables 2-1 and 2-4. The top line in each table provides ideal user requirements utilized in this study. Tables for all concepts and user applications are included in Volume II.

The critical performance parameters for most of the concepts considered are: effective aperture area, the beamwidths at lowest practical frequency, beamwidths of the device (if applicable), and an estimate of the largest critical structural deviation. Since most of the large structure candidates evaluated were electromagnetic radiation energy collectors, these particular performance parameters apply.

2.8.2 RDT&E. Significant problems associated with the development of large, lightweight structures were identified. Most of the problems are related to the extreme size of the experiment concepts and the mechanical tolerances required for satisfactory operational performance. Some of these are:

a. Packaging and deployment.

b. Joining and assembly with EVA.

c. Alignment.

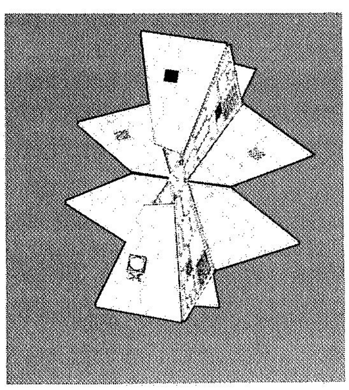
d. Thermal distortions.

e. Rigidity and maneuverability.

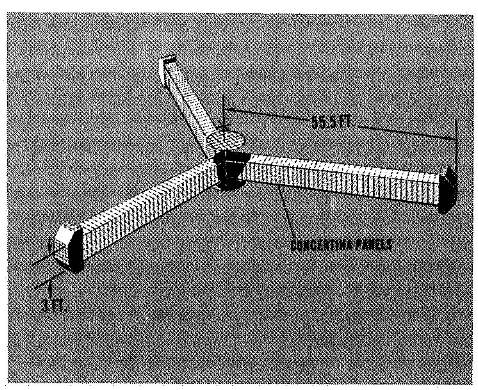
f. Ground testing and checkout.

A summary of the typical engineering tests required for each concept is given in Volume II.

DEEP SPACE MICROMETEOROID PANEL
Concept 14 - Martin



CONCERTINA MICROMETEOROID
Concept 18 - Fairchild/Hiller




METEOROID COLLECTOR
Concept 85 - Convair

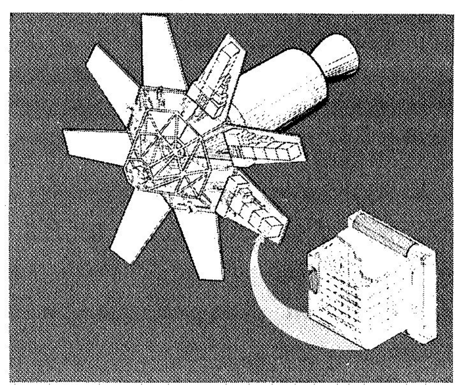


Figure 2-7

2.8.3 COST AND SCHEDULES. An abbreviated cost and schedule analysis was undertaken for each of the concepts. Results are recorded in Volume II.

## 2.9 CONCEPT SELECTION

Subsequent to the midterm presentation, NASA selected the following three large space structures for more detailed analysis and design during the second half of the study:

Crossed H Interferometer. This concept (essentially the same as Concept 51b) was to be capable of accomplishing a wide variety of long wave radio astronomy observations in the 0.5 to 10 MHz range, and is assumed to be launched by a manned Saturn V launch vehicle into a synchronous orbit. Results of the detailed preliminary design and analysis of this structure are contained in Volume III. A summary is presented in Section 3 of this report.

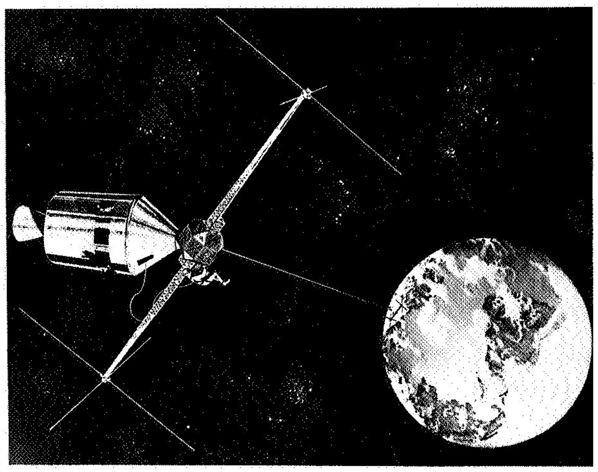
Focusing X-Ray Telescope. The second selected concept was a focusing x-ray telescope very similar to Concept 73, but smaller in size. It was NASA's assessment that the appropriate size of such an instrument for flight in the mid-1970s should be in the neighborhood of 20 - 40 inches. Thus, the preliminary design and analysis of a "nominal 30-inch aperture" device was established as a guideline for the second half of the study. It was further directed by NASA that the x-ray telescope should be capable of accommodating a wide variety of soft x-ray observations throughout its useful lifetime. Detailed results of the subsequent study effort on the x-ray telescope are found in Volume IV; a summary is presented in Section 4 of this report.

100-Foot Parabolic Antenna. The third concept selected by NASA is a nominal 100 ft diameter parabolic antenna similar to Concept 102. Although no operational mission is directly associated with this selected concept, the parabolic dish is considered a technology step leading to the development of the structures technology required to accomplish a broad spectrum of potential future missions such as point-to-point communications and TV broadcast. Results of the preliminary design of this concept are contained in Volume V; a summary is presented in Section 5 of this report.

It was further directed that the preliminary design and analysis of these three concepts be pursued with the three primary flight objectives in mind: (1) the evaluation of man's role in the assembly, alignment, checkout, etc., of a large structure in space, (2) gaining structures technology, and (3) as appropriate to suit the particular mission.

# SECTION 3

## CROSSED H INTERFEROMETER FOR LONG WAVE RADIO ASTRONOMY



### 3.1 INTRODUCTION

This section presents a summary of the results of the study of the crossed H interferometer for long wave radio astronomy. Basic ground rules were:

a. Saturn V Manned Launch

b. Circular Synchronous Orbit, 28.5° Inclination

c. Mission Lifetime One Year Minimum

d. Manned Resupply/Repair Flight, if required

e. Orbital Crew Activities Constrained to the MSC "Baseline Extravehicular Astronaut 1968 to 1972"

The study was primarily concerned with engineering problems in the following areas:

Structural/Mechanical Design and Analysis
Packaging
Subsystems
Materials
Alignment
Thermal and Dynamic Analysis
Determination of Man's Role in Deployment, Operation and Maintenance of the Facility.

The following principal conclusions have been reached:

a. To the extent of the analysis performed, it is feasible to design, develop, and deploy the crossed-H interferometer facility by 1972, which will provide radio astronomy observations of 0.5 MHz to 10 MHz.

b. The facility can be placed in synchronous orbit by a manned Saturn V launch, with adequate reserve payload weight and volume for growth and additional experiments.

c. The structural concept provides an excellent facility for the demonstration of the operation and performance of extendible booms and extendible tubular mesh structures under various thermal and dynamic environments. The dynamic behavior of tethered, gravity gradient stabilized bodies may also be evaluated.

d. Man's primary roles would be to (1) monitor the deployment and checkout while providing a manual backup to the automatic system, and (2) subsequently provide orbital maintenance and refurbishment. Through photography and medical sensors, the astronaut's dexterity and physical performance may be recorded to validate his ability to function as a valuable adjunct to the orbiting structure.

## 3.2 FLIGHT OBJECTIVES

One of the primary flight objectives of the crossed-H antenna is to verify the use of man in erecting and maintaining the operation of a large orbiting structure.

To achieve this goal on a justifiable basis, the astronaut's capabilities were weighed against automation on the system level and redundancy on the component part level. Such factors as reliability, work hazards, costs, and EVA work constraints were used as a basis for analysis. This analysis indicated that man's activities were indispensable in assuring mission success in the areas of:

a. Initial Deployment and Checkout

b. Malfunction Repair

c. Scheduled Refurbishment

While accomplishing the above tasks, the astronaut will furnish valuable data for future orbital work through the media of photography and biomedical sensors.

Through the use of photography, the astronaut's abilities to do the assigned tasks may be recorded. His ability to handle and replace various size components may be confirmed, from the fairly large solar cell panels to the small attitude control jet modules. His methods of locomotion will also furnish valuable data upon which future activity may be based. Such equipment transportation methods as the "clothesline" principle may be observed for future applications.

The advancement of large orbiting structural technology is one of the prime flight objectives of the crossed-H antenna. Through the use of thermal sensors, strain gauges, and photography, the structural behavior may be telemetered to earth for analysis and subsequent structural advancement.

A most significant area of technological development is the tether. Studies have been performed on tether dynamics and it is anticipated that experience with orbital tethers will be gained before the launch of this experiment. But it is probable that the use of this tether length in repeated extension and retraction will, with suitable instrumentation, provide valuable data on such a typical space structure.

The extendible boom is another example of the application of structural technology to space environment. Normal webbed truss theory is extended to provide a boom with a high degree of immunity to thermal and dynamic distortions while incorporating a repeatable extension ratio of 8 to 1. Strain gauges, thermocouples, and position sensors will provide data to evaluate the effectiveness of the design and lead to new improvements.

Other features of the antenna assembly provide sources of data that will establish and promote structural technology. The dipoles of bi-metallic mesh in the shape of extendibel and retractable tubes should provide additional data in this application, although we anticipate that the fundamental concepts will be proven in preceding experiments or pre-orbital tests.

Although the various drive mechanisms operate normally on the earth, they will be further tested in space by this experiment, particularly with respect to longevity and reliability. The need for such proof is reflected in the incorporation of the detail design of redundancy and provisions for EVA replacement by means of modules and gross attachments.

In short, this design represents a logical and practical extension of structural technology to space applications.

This antenna assembly has been designed to incorporate the scientific equipment that makes possible the investigations which various investigators may wish to perform. The scientific objectives include:

a. Survey the low frequency radiation over the entire sky, with good resolution, including spectral and polarization measurements.

b. Survey low frequency discrete radio sources, with good resolution, including spectral and polarization measurements.

c. Obtain spectral and polarization measurements of the sun, with good temporal resolution.

d. Obtain low frequency observations of Jupiter, and possibly other planetary sources, with good temporal resolution.

It is the intent of the crossed-H interferometer design to use aperture synthesis techniques to perform the investigations under "a" and "b" above; "c" and "d" above are carried out in the frequency range from 0.5 MHz to 5 MHz simultaneously, and in both polarization orientations simultaneously.

## 3.3 ANTENNA PERFORMANCE

The principal measurements desired of a long wave radio astronomy system are (1) the spectral brightness and polarization mapping of essentially time-stationary sources for frequencies below 10 MHz and (2) the spectral brightness and polarization monitoring of strong time-varying sources within the solar system for frequencies below 10 MHz.

Earth-based radio telescopes are limited in their usefulness in varying degrees below roughly 20 MHz by the reflection, absorption, refraction, and polarization rotation effects of the ionosphere. They are also adversely affected by interference from man-made signals and atmospheric noises originating on the earth. These limitations increase in severity with decreasing frequency, becoming very severe at about 10 MHz and intolerable at frequencies below about 5 MHz. Spaceborne antennas operating outside the ionosphere blanket avoid many of the problems associated with earth-based telescopes. Below frequencies of 4 or 5 MHz, spaceborne antennas are the sole means of obtaining measurement data.

To be useful, a satelliteborne long wave radio astromony antenna must, among other things, be able to operate at frequencies below 5 MHz, must be able to resolve to small angles for mapping, must be capable of monitoring time-varying sources, and be able to measure the polarization of incident radiation.

The evaluation of various satellite structural concepts showed that the variable geometry, crossed-H interferometer concept best satisfies the long wave radio astromony user requirements when considering the limitations imposed by structural and dynamic factors inherent in each concept.

A feature which makes the interferometer particularly useful is its variable tether length. This makes possible use of the interferometer, supported by data correlation processes, for an unambiguous mapping resolution equivalent to that of a two-dimensional, filled aperture array. This achieves a performance that could be matched, using conventional techniques, only by a vastly more complex antenna structure. Another very advantageous feature is the variable dipole spacing and length, which permits operation over the broad frequency range from 0.5 MHz to 10 MHz. A third important feature is the ability to slew the end-fire dipole assemblies to continuously monitor one sector of the sky. This permits the use of the instrument to study time-varying sources, such as the sun, when events of special interest occur.

## 3.4 CONFIGURATION DESCRIPTION

3.4.1 OPERATIONAL DESCRIPTION. The crossed-H interferometer long wave radio astronomy antenna is deployed in 28 1/2° synchronous orbit by a manned Saturn V launch vehicle system. The sequence of mission operations is shown in Figure 3-1.

BASELINE MISSION OPERATIONS

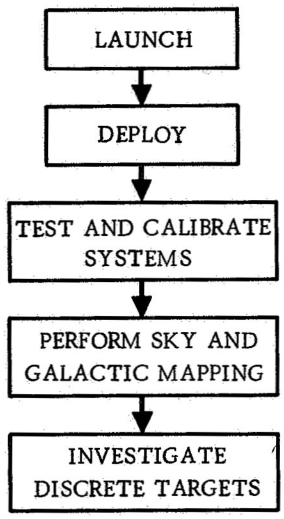


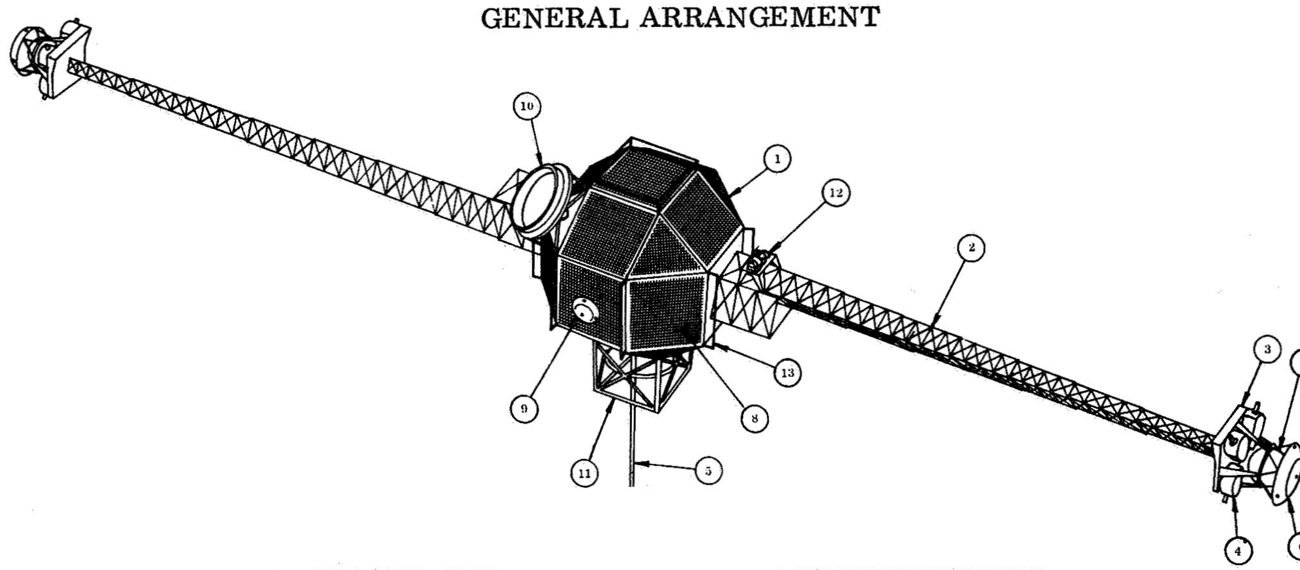

Figure 3-1

The antenna system consists of two satellites which are stabilized in a gravity gradient field utilizing an adjustable length tether. The two satellites are geometrically identical and vary only within the electronic receiving and transmitting networks contained within the centerbooms. Each satellite consists of five major components: the center body, two extendible boom assemblies, and two dipole head assemblies (Figure 3-2).

To achieve the broad performance range exhibited by the antenna the basic geometry of the antenna has been made adjustable. The tether system is adjustable from 0 to 10,000 meters in length. The telescoping booms that support the dipole heads are adjustable from 3 meters to a maximum of 30 meters, while the individual half dipole lengths are variable from 6 meters to 75 meters. See Figure 3-3.

The entire antenna array may be retracted to the launch configuration for gross EVA maintenance and refurbishment functions.

CROSSED H INTERFEROMETER ANTENNA
GENERAL ARRANGEMENT




1. MAIN BODY ASSEMBLY - ACS SYSTEM
   TETHER REEL AND TENSION SENSORS
   RECEIVING AND TRANSMITTING ELECTRONICS
   GROUND DATA LINK ELECTRONICS
   STAR TRACKING SYSTEM
   GUIDANCE AND CONTROL ELECTRONICS
   BATTERY AND CHARGING SYSTEM
   BOOM ACTUATION MOTOR
   STRUCTURAL SENSORS AND TELEMETRY
2. TRIANGULAR TELESCOPING BOOM - 1 FIXED, 7 MOVING SECTIONS
3. DIPOLE HEAD ASSEMBLY
4. DEPLOYABLE MESH DIPOLE UNIT (8)
5. TETHER TAPE, ADJUSTABLE 0 - 10,000 METERS
6. ATTITUDE CONTROL JET MODULE (9)
7. ATTITUDE CONTROL PROPELLANT
8. SOLAR CELL PANELS - 115 SQ FT
9. ATTITUDE CONTROL JET MODULE (2)
10. CSM SATELLITE DOCKING RING
11. INTER-SATELLITE DOCKING STRUCTURE
12. STAR TRACKER OPTICS (2)
13. EVA RESTRAINT RAILS

Figure 3-2

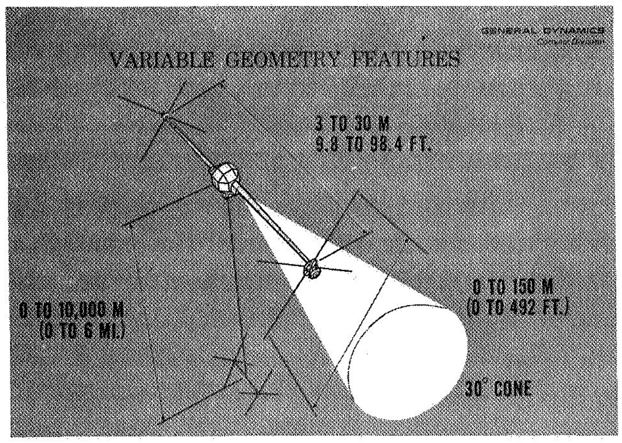



Figure 3-3

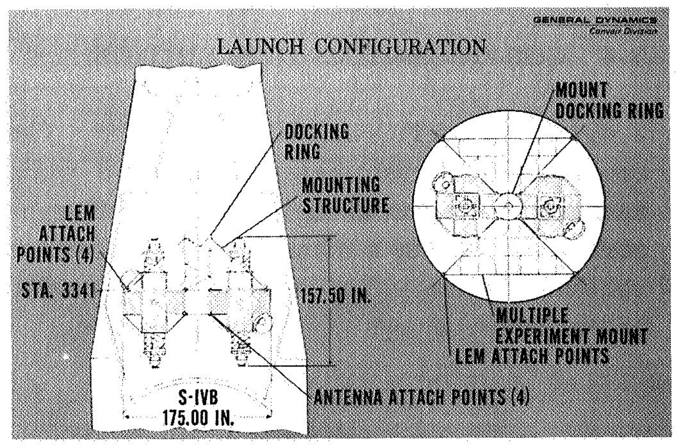



Figure 3-4

Launch Configuration. The antenna is mounted within the SLA payload area of the Saturn V manned vehicle system utilizing the four existing LEM attach points at Station 3341 (Figure 3-4). No modification of the launch vehicle structure is required. The antenna is packaged into a volume of 1170 cu ft or about 28% of the available payload volume. The 4100 lb experiment uses 17% of the allowable payload weight.

The mounting structure is also utilized for the mounting of additional experiments. The entire mount is released from the LEM attach point by four pyrotechnic latches. The crossed-H antenna is retained in the mount by a system of eight pyrotechnic latches.

Deployment. Figure 3-5 shows the orbital deployment of the antenna into synchronous orbit. In the figure the main deployment sequences are illustrated starting with the separation of the CSM and its docking operation to the payload mounting structures. Following the removal of the payload from the antenna assembly using the CSM propulsion system, a complete structural and subsystems checkout is initiated. The subsystems checkout is completed while the antenna is retained on the payload mount. The structural operational check is completed after ejection from the payload mount.

A gross orientation with respect to the desired gravity gradient position is accomplished prior to the antenna separation from the mounting structure. The tether mechanism and attitude control system check is then made by deploying the satellites to a separation distance of 200 meters. At this distance the boom and dipole activation systems are extended and retracted prior to antenna deployment to the maximum 10,000 meter tether length. All checkout and deployment operations are initiated and monitored by the astronauts from within the CSM.

The deployment of the antenna is initiated remotely by the astronaut from within the CSM, thus permitting observation of the deployment and its termination, if necessary.

Antenna deployment is automatic with manned initiation and control; the CSM remains in the vicinity of the antenna during the systems checkout phase and the initial 14 to 28 days of operation so that any early malfunctions may be corrected by the astronaut prior to his departure from orbit.

Orbital Configuration. To gather the desired scientific information the structural geometry of the antenna must be varied as shown in Figures 3-6 and 3-7.

During the 333 days of the source survey mission the tether length is gradually retracted 4 times while the booms and dipoles assume a discrete setting for each tether retraction cycle.

ORBITAL DEPLOYMENT SEQUENCE

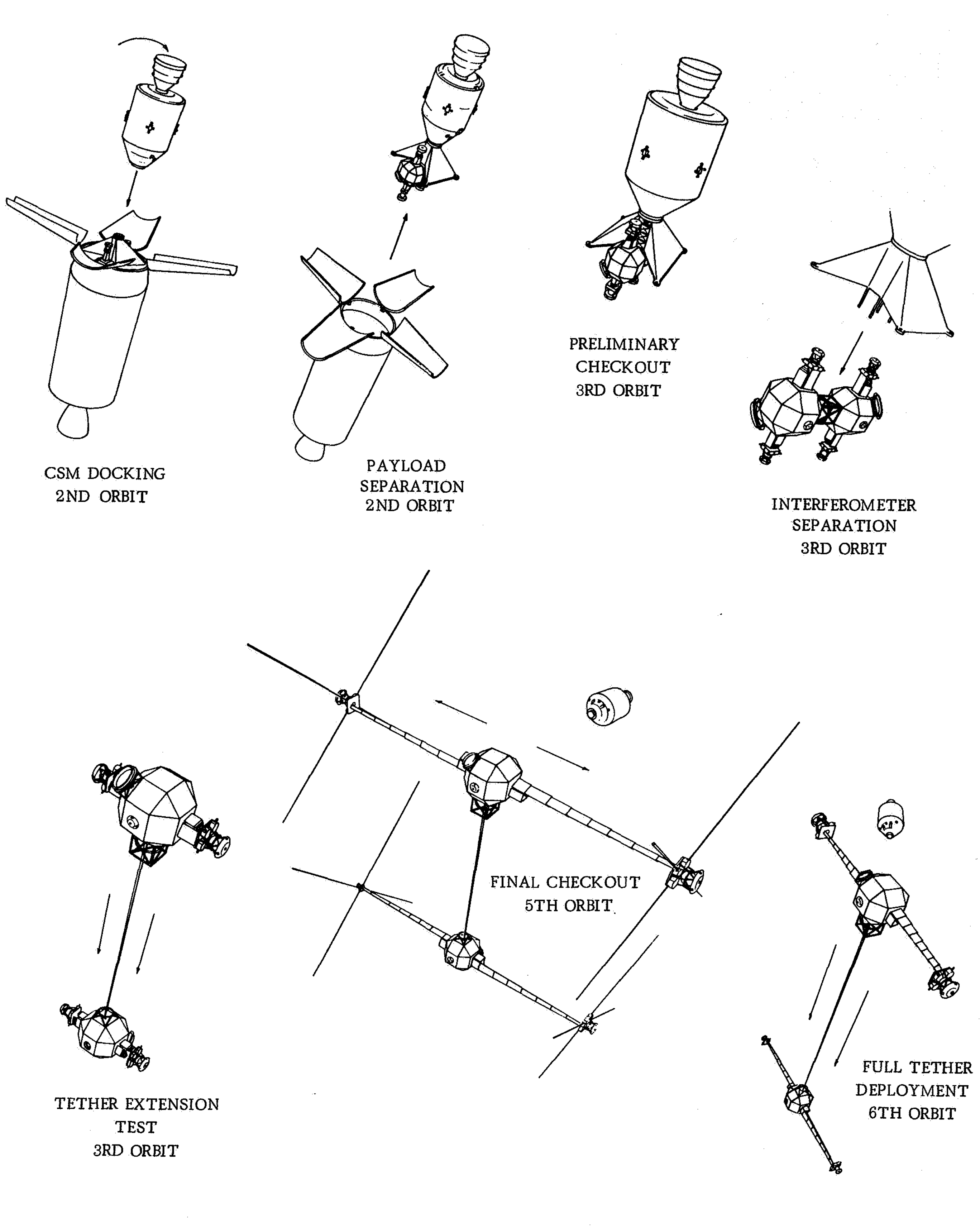



Figure 3-5

## BASELINE SOURCE SURVEY MISSION

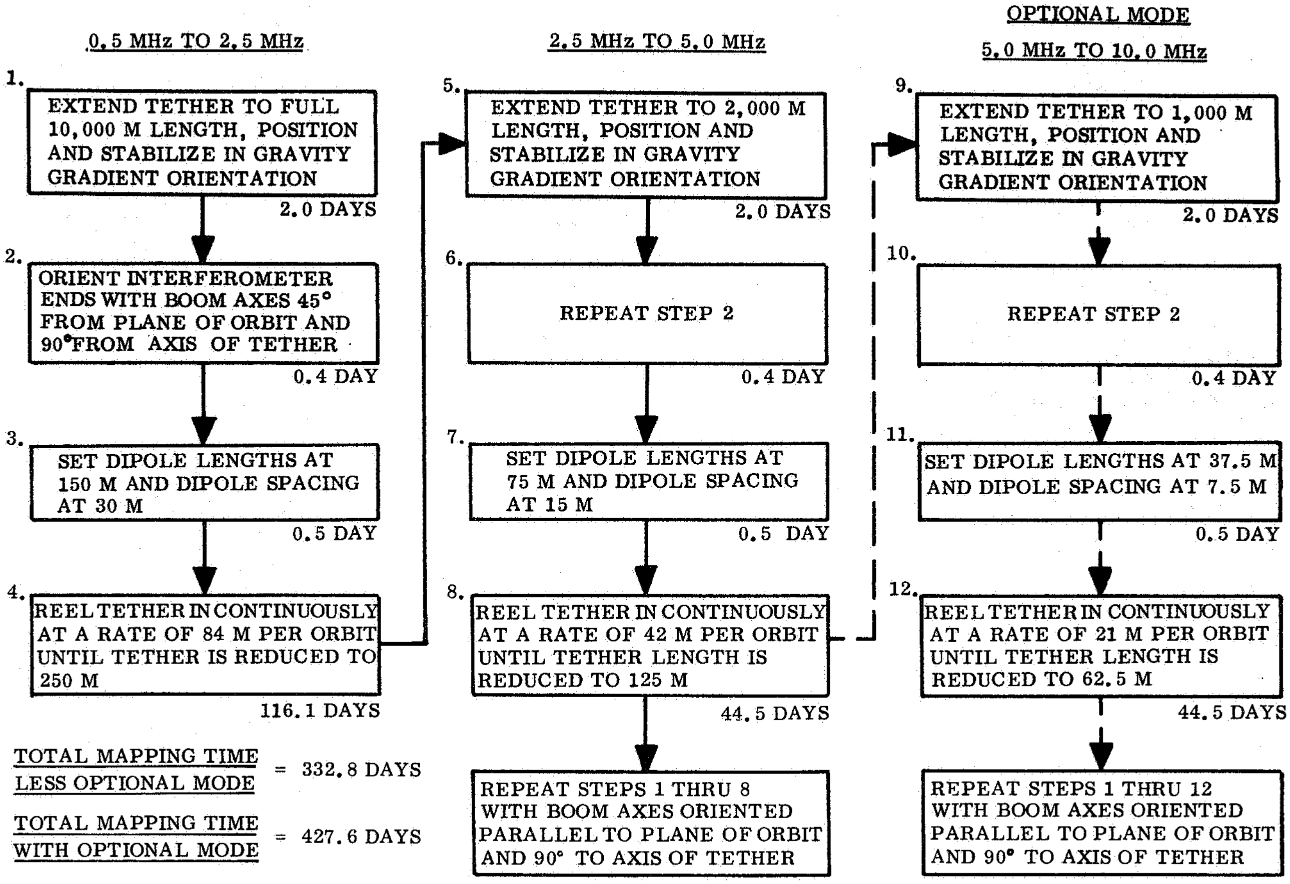



Figure 3-6

STRONG TIME VARYING SOURCE
OBSERVATION MODE

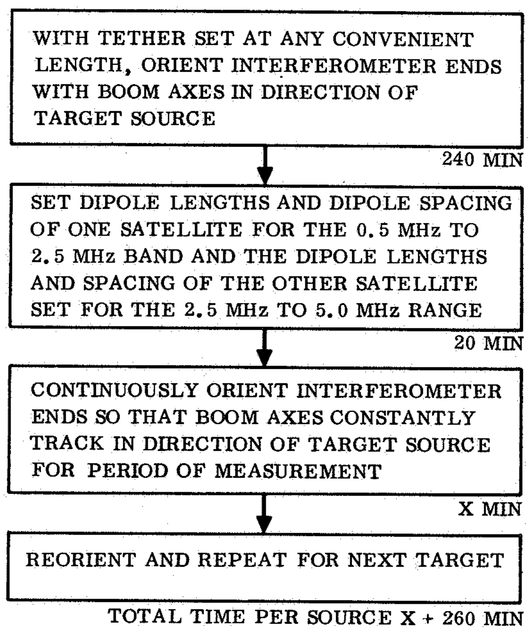



Figure 3-7

Much of the time alloted for maneuvering and structural variation is spent in waiting for damping of induced dynamic oscillation of the main booms. The dipoles are retracted during adjustment cycles of the tether and boom. The average traverse time from source to source during the discrete observation mode is 10-20 minutes. Lockon time is 10 minutes for each desired structural adjustment.

Only one maneuver is required during the mapping mode portion of the mission. This is a rolling motion about the Z or tether axis. This maneuvering about the Z axis provides two orientations of the boom axis, one 90° parallel to the orbital plane and the other 45° to the orbital plane.

After the celestial mapping has been completed, the antenna is utilized for time varying source observation. For this mode the antenna is again rotated about the tether axis, as well as pitched about the Y axis, which is also perpendicular to the tether axis (Figure 3-8). This maneuvering capability provides for extended lockon time, a much greater latitude of time during which a target may be read. For the discrete source observation mode, the satellites are mechanicaly set for two individual frequencies: 0.5 to 2.5 MHz and 2.5 to 5.0 MHz.

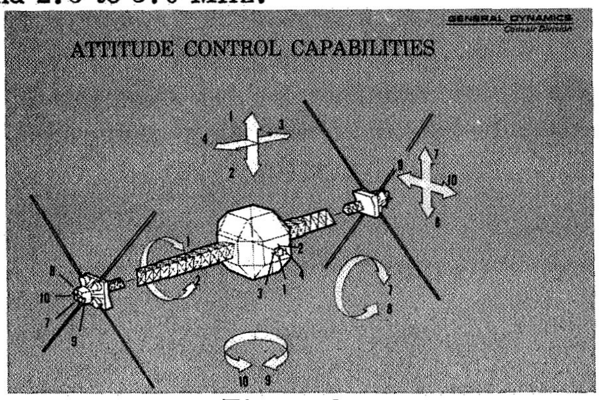



Figure 3-8

3.4.2 STRUCTURAL MECHANICAL DESIGN. The final structural design is the result of the modifications imposed on an optimum antenna configuration by such constraints as thermal and dynamic distortion, systems reliability, and the modifications required to optimize man's role in the mission. A weight summary is provided in Figure 3-9.

The structural/mechanical design effort was concentrated on four major components: (1) the centerbody, (2) the deployment booms, (3) the dipole head assemblies, and (4) the tether and its deployment mechanisms.

GENERAL DYNAMICS Convair Division

WEIGHT SUMMARY

| | SATELLITE I (SLAVE) | | SATELLITE II (MASTER) |
|---|---|---|---|
| * STRUCTURE | 546 | | 546 |
| ACS | 244 | | 244 |
| POWER | 359 | | 359 |
| ** ELECTRONICS | 267 | | 334 |
| EVA PROVISIONS | 212 | | 212 |
| TETHER ASSEMBLY | 56 | | 56 |
| | 1,684 | | 1,751 |
| CONTINGENCY | 316 | | 350 |
| 10% STRUCTURE | | | |
| 50% ELECTRONICS | | | |
| TOTAL WEIGHT | 2,000 | 4,101 LB. | 2,101 |

*DOES NOT INCLUDE 400 LB. MOUNTING STRUCTURE
**DIFFERENCE IS IN RADIOMETRY & DATA LINK SYSTEMS

Figure 3-9

The primary dictates for the geometrical shape of the center body are the satellite power requirements and the resulting solar cell area and arrangement. The centerbody configuration is a 26-sided figure octagonal in cross section. This results in a surface which is composed of eighteen 30 x 30 in. square panels and eight 30 in. equilateral triangular panels. Fifteen of the square panels and seven of the triangular panels are used for the solar cell array. The solar cell panels are of modular design to provide interchangeability for malfunction repair and maintenance by the astronaut. Defective cell modules are not removed from the satellite but are overlaid by the new panel. The replacement panel snaps into place and requires no fasteners. Electrical connection is made automatically as the panel is snapped in place.

Figure 3-10 shows the general center body structural and equipment arrangement. Satellites 1 and 2 are identical except for a few minor differences in the electronic subsystems. Enclosed within the centerbody are the various electronic subsystems such as the data receiving and transmitting systems, telemetry, autopilot, and star tracking systems. Also within the centerbody are the two fixed deployment boom sections from which the extendible sections are deployed. The tether deployment and length counter system is duplicated within the centerbody of both satellites for increased reliability since it would be extremely difficult to repair or replace by EVA. Ten thousand meters of tether are provided, 5000 meters on each reel. The tether is replaceable by the astronaut.

The boom deployment motor is also within the center module. A single package houses a dual motor, one of which is redundant. The single dual motor module provides for both the extension and retraction of both booms. The motor module is replaceable by EVA. Access is through a hinged solar cell modular panel. The spent unit is removed and the new unit clamped in place without the use of

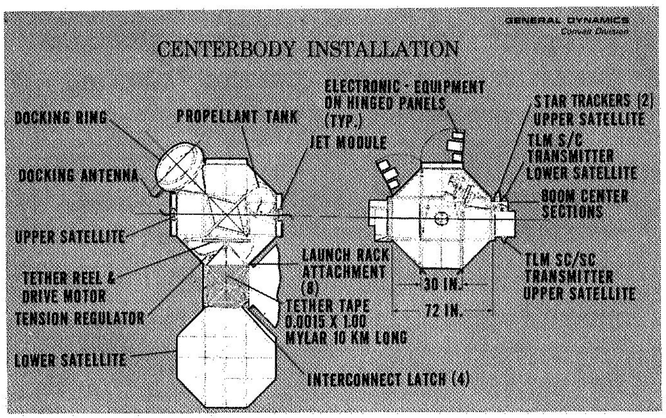


Figure 3-10

fasteners or special tools. Also contained within the centerbody is the attitude control system required for tether extension and retraction and roll about the Y axis and translation along the X axis (see Figure 3-8). The cold gas propellant ($N_2$) tankage for these jets is also within the centerbody. The tank may be refilled by the astronaut by connection to an external fitting.

A docking ring is provided on both the satellites thus providing docking capability for the CSM during any EVA activity. The booms and dipole elements are completely retracted prior to the docking operation, thus reducing the possibility of structural damage. EVA distances are also reduced to a minimum.

The extendible boom structure is composed of seven sections and one fixed section (Figure 3-11). The sections are triangular in cross section. The side webs of each section are built up of aluminum alloy sheet metal and are riveted to the extruded aluminum corner cap members.

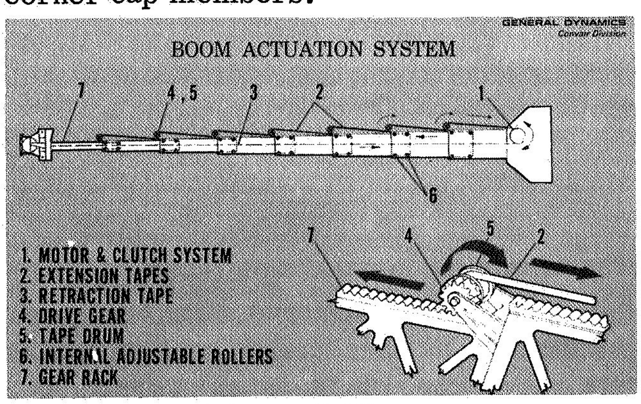


Figure 3-11

The moving sections are extended by a gear and rack system. The gear is actuated by a tape drive. Retraction is done by a single tape running through guides on the boom. Rollers at all three corners run on tracks provided on the inside of the corner cap members of the preceding member.

The dipole head assembly consists of an equipment box, dipole actuation motors, dipole deployment units, and the attitude control system required for yaw about the tether axis and pitch and translation about and along the Y axis (Figure 3-12).

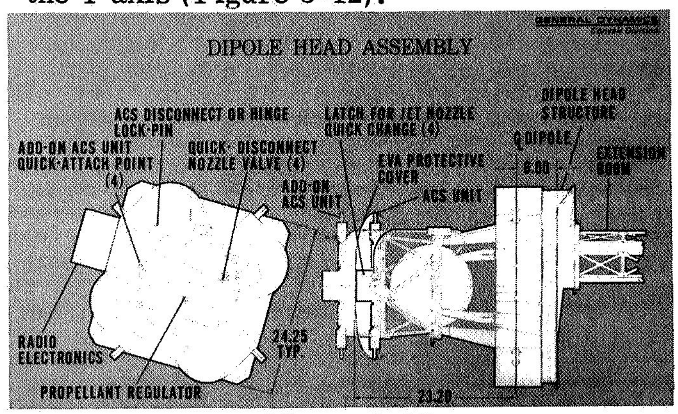


Figure 3-12

Provisions are made for complete head assembly overhaul at the time of gross orbital refurbishment by the astronaut. Design of the head assembly provides for repair and refurbishment by modular replacement for the dipole drive motor, the dipole deployment units, and the attitude control module, which contains the jets, control valves, and pressure regulator assembly. A faulty dipole deployment unit is replaced by unlatching the unit and rotating it 90° on its hinged base and then plugging in the new unit utilizing the same snap-on fasteners. The attitude control system module is repaired by plugging in a new unit on top of the faulty assembly; pressure and propellant is automatically transferred to the new unit. The dipole deployment motor unit is replaceable by swinging the faulty unit aside and inserting the new unit.

The attitude control propellant is replenished by the same method described for the center body supply.

The tether system consists of the tether tape, deployment reels, tension sensors, deployment motors, and intra-satellite distance and position sensing equipment. See Figure 3-13.

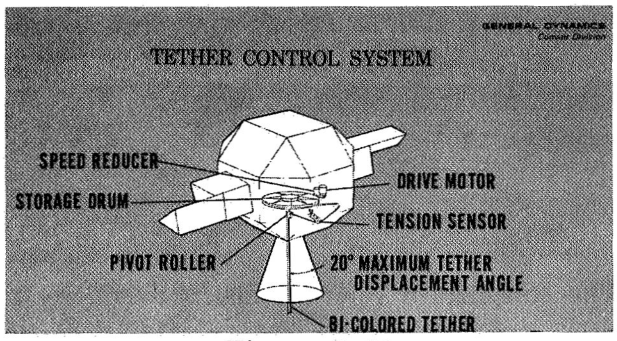


Figure 3-13

The problem of tether severing by meteoroid impact is minimized by using a flat tape of 1/2 mil by 1.00 inch mylar. Solar pressure and the resulting bowing of the tether is minimized by offsetting the center of pressure and center of cross sectional mass through the use of a bi-colored tape. This produces a "weather vaning" effect on the tether.

The deployment reel and tension sensing system is duplicated in each satellite, thereby furnishing a backup system should either system fail. The intra-satellite distance sensing equipment is used to activate the attitude control system, which provides the required impulse for tether stability. A single motor is provided in each satellite so that the drive actuation system is also redundant.

3.4.3 DYNAMICS. The dynamic behavior of this concept is anticipated to be satisfactory. More detailed analyses are required in two areas to support the preliminary work. These areas are tether-satellite behavior and docking. Active attitude control of each satellite is provided, making possible both source lockon observation and scan orientations in other than the gravity gradient stable position.

The major problem inherent in the dynamics and attitude control technology area for the crossed-H concept as envisioned in this study is best stated as a credibility problem. This is the dynamic behavior of the tether and the two satellites, particularly during the slow separation change operation. It is believed that the presence of active control of each satellite and the computer will allow control of the structure to the desired dynamic behavior. This assumption will have to be the subject of a detailed simulation during future work.

Docking dynamics have not been examined. Since the individual satellite is small compared to the CSM, the docking impulse imparted to the satellite, particularly for a non-latch contact, may cause appreciable motion. Satellite deflection will be minimized by shock absorbers and retraction of booms and dipoles during docking.

The dynamic behavior of the crossed-H dipoles are of primary concern. Estimates of their dynamic characteristics show that the dipole oscillations will be within acceptable limits. Boom oscillations should be smaller.

Each of the two satellites have nearly identical attitude control systems. These systems maintain the desired orientation and, acting together, control the motion of the complete experiment.

An all-jet $N_2$ system has been selected. Inclusion of inerita wheels would not save weight, and reliability improvement has not been demonstrated. Propellant selection of $N_2$ was made on the basis of higher reliability with acceptable associated weight and volume penalties.

Jet reliability is extremely important. Some of the jets have functions which can be supplied by other jets, at the expense of lighter-than-planned propellant consumption from a particular tank. Some of the jets have functions which cannot be performed by other jets, in which case the experiment is discontinued until they are repaired.

3.4.4 THERMODYNAMICS. Thermal distortion was computed for the extending boom and for the dipole elements. Computer solutions were employed due to the complex geometry and associated conduction/radiation heat flow.

Results revealed that boom distortion could be reduced by nearly a factor of four by utilizing white paint (to obtain low absorptance and high emittance) and by staggering the web pattern on the boom faces (to allow internal illumination). A total angular displacement of 2.2 degrees was calculated for the 30-meter boom.

It is recommended that the dipole elements be constructed of wire screen material to minimize distortion. A good thermal conductor, beryllium/copper, is used for the circumferential wires while the thermal expansion is reduced by using elgiloy for the longitudinal wire. Furthermore, the use of silver plating was shown to reduce deflection by more than a factor of five. Distortion is not sensitive to wire diameter but is nearly linearly proportional to the mesh size. Analysis of the final dipole design resulted in maximum tip deflection of about 4.5 meters for the 1.5-inch-diameter tube fully extended to 75 meter length. See Figure 3-14.

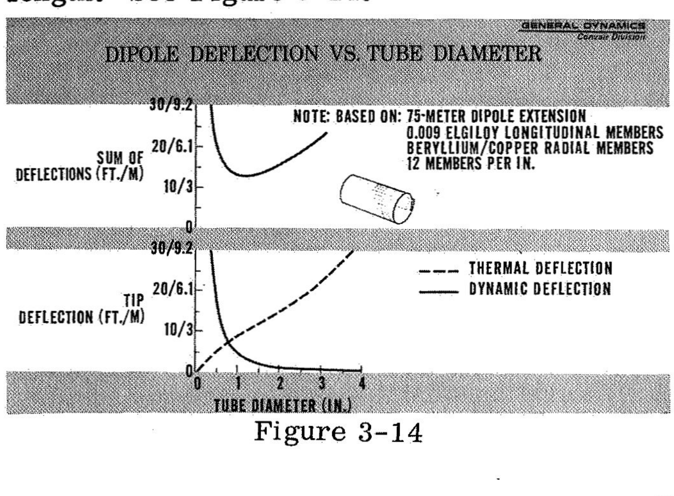


Figure 3-14

Subsystem orbital environment was determined by evaluating the thermal balance on the main body housing including internal power dissipation. While in the sun, the average temperature within the mainbody will be between 40° F and 66° F. Cooling occurs during transit through the earth's shadow. It was estimated that the equipment would emerge at about 30° F.

3.5 EVA PARTICIPATION

The function of man has been analyzed and reflected in the basic design of the vehicle to provide not only the greatest possible assurance of mission success, but also to show man's capability to perform the EVA tasks necessary for successful operation. Man's role in the overall mission profile is to serve in the areas of deployment observation, check out, malfunction repair, and gross refurbishment (Figure 3-15).

The basic satellite systems and component parts have been analyzed with respect to expected failure rates, astronaut capabilities, hazards and overall system cost. This analysis serves as the basis on which the operational mode of each system is selected. A component part failure analysis is used to determine whether the part should be redundant or replaceable by the astronaut. Using this approach, man's activities have been designed into the mission only when justified on the basis of increased probability of mission success.

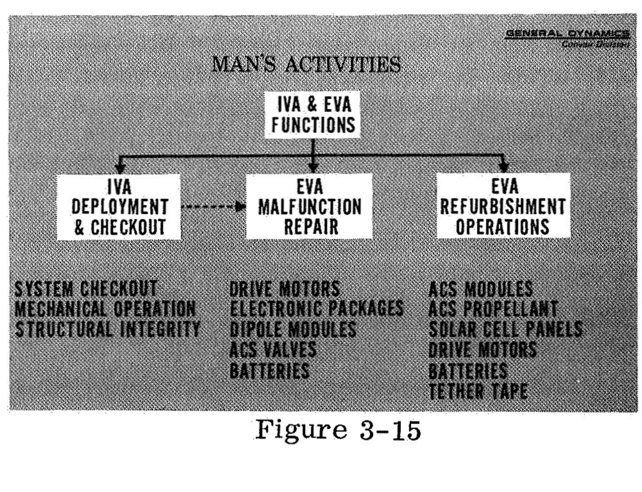


Figure 3-15

All equipment replaceable by EVA has been designed to be removed and replaced by simple pull pin or lever action mount. The electrical and mechanical interfaces are generally completed in one simple step.

The CSM is docked to the satellite for all maintenance and refurbishment work which is done by EVA. The booms and dipoles are retracted to minimize the EVA work area; this allows all work to be done using a 50-foot tether. See Figure 3-16.

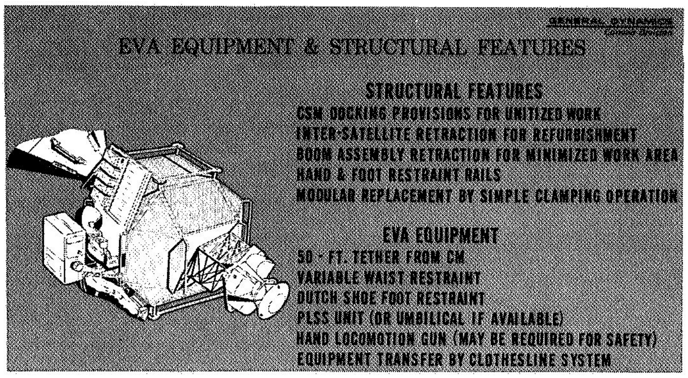


Figure 3-16

## 3.6 RDT&E

The purpose of the research, development. test, and engineering plan is to provide the planning documents and information which define all steps required to achieve a functioning orbital antenna. The following are included in Volume III for the crossed-H interferometer: (1) work breakdown structure; (2) prerequisite orbital experiments; (3) research, manufacturing, test, and support plans; (4) schedule; and (5) cost analysis.

Only one new facility is identifiable at this time: a zero-g deployment facility. Deployment of the telescoping truss booms in the horizontal position will require an overhead track system which will support the assembly by negator springs. An alternate approach would require a 60-foot vertical tower with a pulley-counter-weight system for upward extension of the boom.

Prerequisite orbital experiments which (1) establish EVA locomotion loads on the structure and (2) demonstrate the practicability of tool and equipment transfer to the remotely positioned EVA astronaut would be desirable and increase the probability of accomplishment of the mission. Two additional areas in which technological advancement would support this program are in the behavior of tethered, gravity gradient stabilized satellites in synchronous orbit and in the dynamic/thermal behavior of long, extendible elements. These may be verified prior to orbit of the crossed-H interferometer by planned and proposed orbital tests by GSFC, such as the radio astronomy experiments and the tethered orbiting interferometer.

Design of the crossed-H interferometer is based on current state of the art in development of materials and construction details. While not specifically required, research in such areas as the manufacturing techniques for screenboom tubing and dynamic analytical models of complex space structures will contribute to performance, cost, or schedule improvements.

Manufacturing processes required to fabricate components are essentially those processes and techniques familiar to airframe and aerospace manufacturers. The basic facility requirements for fabrication, assembly, and checkout of the structure and various systems consist primarily of conventional cable/electronic package fabrication and test equipment. No unusual problems are envisioned in the test areas.

The schedule summary in Figure 3-17 indicates a five year development program, resulting in a mid-1972 launch.

Funding requirements for the antenna program are summarized in Figure 3-18. This includes one flight experiment vehicle only, with no backup flight article.

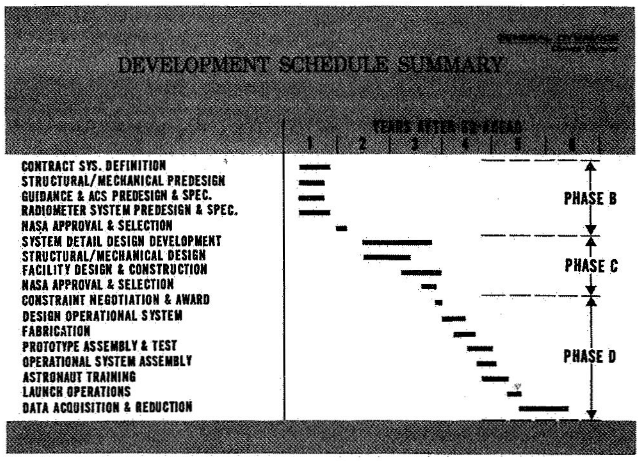


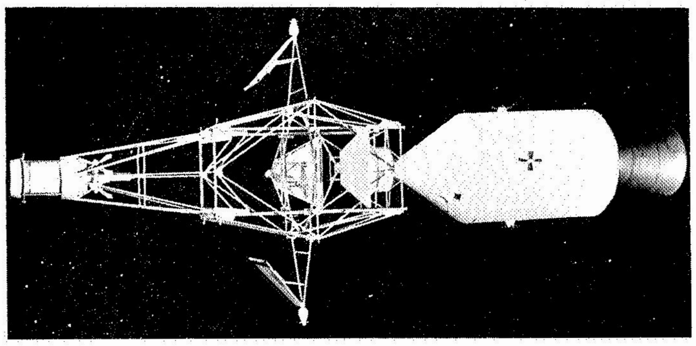

Figure 3-17

**COST ANALYSIS SUMMARY CROSSED-H INTERFEROMETER, LONG-WAVE ASTRONOMY ANTENNA**

| | | $ MILLION |
|---|---|---|
| NONRECURRING | | $20.70 |
| DEVELOPMENT | $19.80 | |
| GSE/STE | 0.80 | |
| FACILITIES | 0.10 | |
| RECURRING (UNIT COST) | | 7.30 |
| TOTAL SYSTEM COST | | $28.00 |

Figure 3-18

# SECTION 4
# FOCUSING X-RAY TELESCOPE



## 4.1 INTRODUCTION

This section summarizes the results of the study of a large orbital focusing x-ray telescope.

The basic study ground rules used were:

a. S-IB Manned Launch around 1974 - 1975 .

b. Circular 260 n.mi. orbit at $28.5^{\circ}$ inclination or less.

c. Mission lifetime greater than three years.

d. Grazing incidence lens diameter range 20 to 40 inches.

e. Manned resupply/repair flights as required.

f. Orbital crew activities constrained to MSC "Baseline Astronaut 1968 to 1970."

The study emphasized the major engineering problems concerned with the following areas:

Structural Design and Analysis
Mechanical Design
Packaging
Materials
Subsystems
Alignment
Dynamic and Thermal Analysis
Determination of man's role in Deployment, Operation, and Maintenance of the System

The following principal conclusions have been reached:

To the extent of the analysis performed it is feasible to design, develop and deploy a 20- to 40-inch diameter grazing incidence x-ray telescope prior to 1975 which will image soft x-ray sources to 1 to 5 arc seconds resolution and perform spectrographic and polarization measurements.

a. The telescope system can be orbited by a manned Saturn IB with adequate reserve payload weight and volume for growth and additional experiments.

b. Three to four segment nested mirrors in the 20- to 40-inch diameters can be developed within five years after go-ahead; eight to ten segment assemblies will require an additional 1-1/2 years.

c. The primary scheduled role of man is to control and assist initial deployment, biannually resupply expendables, replace short lifetime components, and update the vehicle capability by adding new or redesigned equipment.

d. Man's unscheduled activities will increase mission reliability and operational lifetime by repair and replacement of failed or damaged modules.

e. Telescope structures can be designed with adequate rigidity, deployment reliability, and thermal stability or appropriate thermal distortion compensation systems for the 20- to 40-inch lens diameters; structures for lens diameters greater than approximately 50 inches will require less reliable high expansion ratios to meet the S-IB manned launch envelope. Maximum lens sizes for the unmanned envelope have not been determined. Lens focal-length-to-diameter ratios of 10 to 1 or greater were assumed.

The orbital deployment sequence of the x-ray telescope is shown in Figures 4-1 and 4-2.

## 4.2 FLIGHT OBJECTIVES

The primary flight objectives of the x-ray telescope are threefold:

a. Evaluation of the role of man in the deployment, assembly, alignment, and maintenance of a large space structure.

b. Advancement of structural/mechanical technology by orbiting structure performance evaluation.

c. Provide a structure to fulfill a useful scientific/technical mission.

The evaluation of man's role to provide invaluable date on the capability of man in space will be achieved by standard biomedical data, obtained on a real time basis for maximum crew safety. Man's effectiveness or difficulty in performing the various tasks will be analyzed through the specific heart rate, body temperature, and metabolic expenditure data in addition to voice communication recordings. Photographs or TV of all critical EV activity would be essential to establish the reasons for any failure to accomplish an objective and would contribute to modification of equipment and procedures on future flights. An evaluation of the validity of assumed time spans for task accomplishment will provide particularly significant information for future designs. All procedures will be rehearsed with neutral buoyancy or workshop simulation to verify the task validity and familiarize the astronauts with the tasks. Biomedical data will be recorded in the rehearsals for comparison to the flight data. This will provide invaluable data to guide planning of the advanced missions.

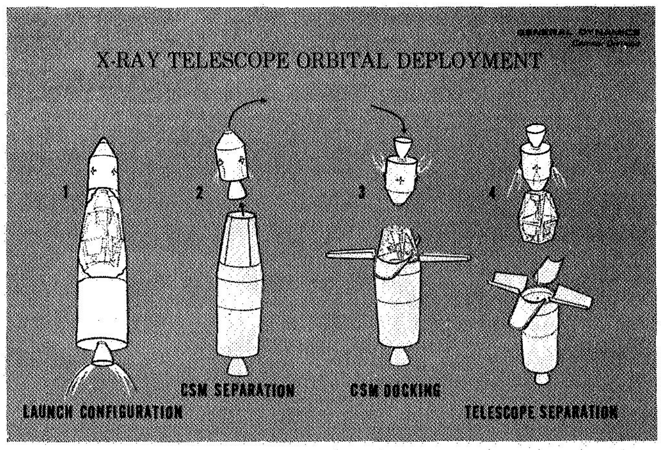



Figure 4-1

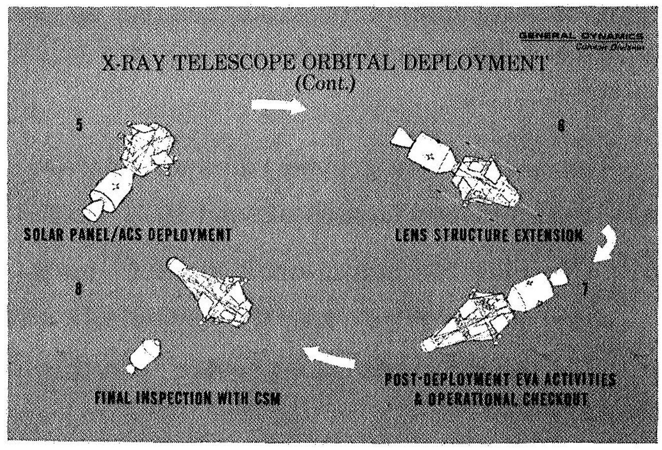



Figure 4-2

The ultimate structural flight objective will be to perform the desired mission with a structure which is an improvement in the state-of-the art in orbiting structures. Since structure behavior in terms of stiffness and thermal stability affect the imaging resolution, the extent of any structure excursions must be monitored to properly evaluate the performance of the mission instrumentation. Conventional instrumentation of strain gages and thermocouples will be used to determine specific member strains and temperatures at strategic points on the structure. The overall optical axis distortion will be computed routinely by an active lens alignment system for use in the navigation system. This time phased data will provide an excellent record of the structure behavior in various orbit positions and will show the effect of the orbit duration on thermal coating.

If further study produces sufficiently thermally stable materials for the structure which would eliminate the lens alignment system, a similar angle and offset laser system can be used to generate the total distortion information. The degradation of bearing or increase in friction of the continuously functioning mechanical systems will be recorded by monitoring drive motor input power; this applies particularly to assemblies such as the equipment turret bearings which are extremely difficult, if not impossible, to replace in the current concept.

The ultimate scientific objectives of stellar x-ray astronomy as summarized by the Wood's Hole report are:

a. Search for discrete sources $10^{-3}$ to $10^{-6}$ times weaker than observed to date.

b. Precise x-ray source location to within 1 arc minute for possible correlation with optical or radio sources.

c. Structural studies of discrete sources with less than 5 arc sec resolution.

d. Spectral distribution studies of discrete sources.

e. Polarization studies of discrete sources.

f. Directional and spectral study of diffuse radiation.

g. Search for intensity time variation of discrete sources.

Solar investigations differ from stellar investigations only in the angular size and intensity of the solar disk and radiation.

The analysis of the nominal 30-inch diameter telescope indicates that the majority of desired scientific goals for the 1970 to 1975 time period can be met except for the lens resolution goal of 1 arc sec. A more realistic manufacturing resolution goal is 2 to 5 arc sec.

## 4.3 TELESCOPE PERFORMANCE

The initial phase of the study involved an extensive investigation of scientific user requirements, the specific measurements, tolerances, and limits which would be necessary for the x-ray facility to satisfy the third flight objective. The user goals summary is compared to the anticipated performance of the telescope design in Figure 4-3. The proposed design is a high resolution device with a narrow field of view and consequently best suited to precise measurements of known or

GENERAL DYNAMICS
Convair Division

USER REQUIREMENT COMPARISON

| PARAMETER | GOALS | ACHIEVABLE |
|---|---|---|
| FIELD OF VIEW | 10' - 30' ARC | 30' ARC |
| RESOLUTION | 2" ARC OR BETTER | 1" TO 5" ARC |
| POINTING CAPABILITY | 5' ARC OFFSET | 5' ARC OFFSET |
| LOCK-ON ACCURACY | WITHIN 1'×1' ARC | 1'×1' ARC |
| JITTER | <1" ARC/SEC. | ≤1" ARC/SEC. |
| LOCK-ON TIME | UP TO 360 MIN. | UP TO 90 MIN. |
| BANDWIDTH | 2 TO 300 Å | 2 TO 300 Å |
| SPECTRAL RESOLUTION | $\lambda/\Delta\lambda$ 100 TO 1,000 | 1,000 |
| POLARIZATION | YES | YES |
| APERTURE | LARGE AS POSSIBLE | 30 IN. NOM. DESIGN PT. FOR STUDY |

Figure 4-3

previously identified sources. The events of a typical observation cycle are shown on Figure 4-4. The comparison shows that the baseline design will satisfy nearly all of the measurement requirements. The proposed 90-minute lock on time can be increased to 360 minutes for a 400 lb vehicle weight penalty.

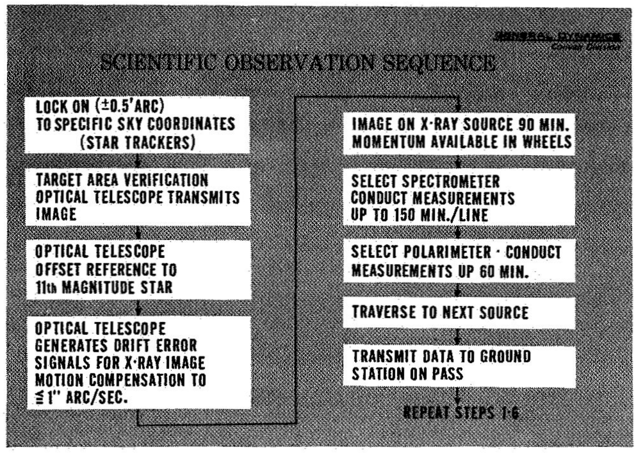



Figure 4-4

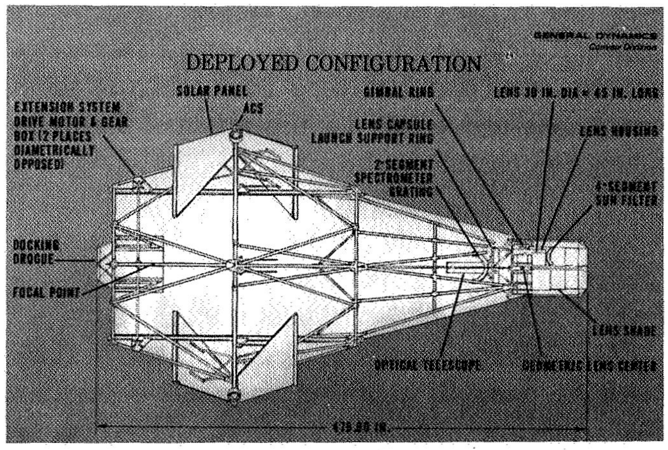



Figure 4-5

## 4.4 CONFIGURATION DESCRIPTION

Selected configuration for the focusing x-ray telescope with a 30-inch nominal lens diameter is shown in Figure 4-5.

4.4.1 PRIMARY STRUCTURE. The basic structure consists of two telescoping truss assemblies fabricated with thin-walled, 2-inch diameter titanium tubes attached to precision cast socket fittings. The 40-foot deployed truss assembly tapers from a maximum depth of 13.3 feet to 8.7 feet on the aft end, and 40 inches on the forward end. This geometry provides maximum structural stiffness consistent with the booster launch envelope. See Figure 4-6.

Preliminary structure concepts evolved octagonal geometries to provide maximum solar heating symmetry for a fully shrouded vehicle. The cross section was changed to a square for simplicity when thermal distortion analysis showed little improvement in the structure distortions with a full shroud. The open structure has the further advantages of maximum access, minimum weight, and minimum aerodynamic drag.

The truss tube size was selected for astronaut hand grip compatibility rather than stress or stiffness requirements, which are very low.

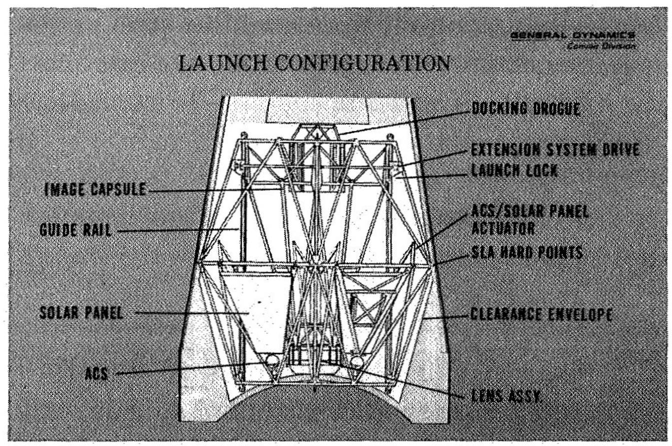
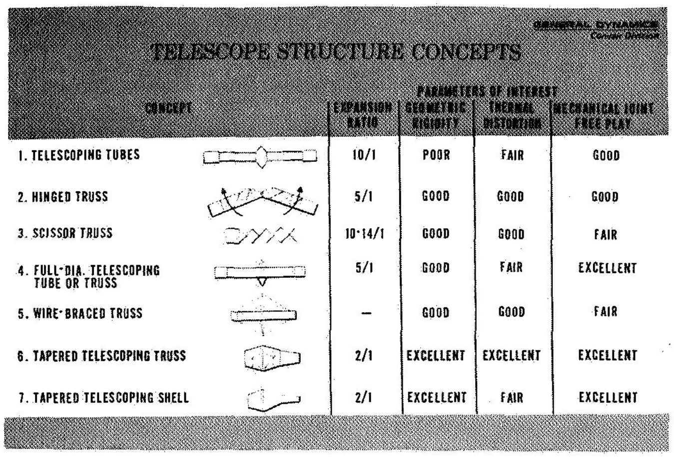



Figure 4-6

The ratio of deployed to stowed structure length is 2/1; therefore, a simple rail-guided slider extension system is used for maximum reliability. The sliding truss sections are positively locked together after extension, thus eliminating all joint slack.

A few of the other structural concepts considered and their advantages and disadvantages are shown on Figure 4-7.

Materials. The x-ray telescope truss material ranks second only to structure geometry in affecting the overall optical axis stability. In addition to the normal structural material properties of interest such as density, modulus of elasticity, and allowable stress, the thermal coefficient of expansion is of particular importance in the design of any telescope



TELESCOPE STRUCTURE CONCEPTS

| CONCEPT | EXPANSION RATIO | GEOMETRIC RIGIDITY | THERMAL DISTORTION | MECHANICAL JOINT FREE PLAY |
|---|---|---|---|---|
| | | PARAMETERS OF INTEREST | | |
| 1. TELESCOPING TUBES | 10/1 | POOR | FAIR | GOOD |
| 2. HINGED TRUSS | 5/1 | GOOD | GOOD | GOOD |
| 3. SCISSOR TRUSS | 10-14/1 | GOOD | GOOD | FAIR |
| 4. FULL-DIA. TELESCOPING TUBE OR TRUSS | 5/1 | GOOD | FAIR | EXCELLENT |
| 5. WIRE-BRACED TRUSS | – | GOOD | GOOD | FAIR |
| 6. TAPERED TELESCOPING TRUSS | 2/1 | EXCELLENT | EXCELLENT | EXCELLENT |
| 7. TAPERED TELESCOPING SHELL | 2/1 | EXCELLENT | FAIR | EXCELLENT |

Figure 4-7

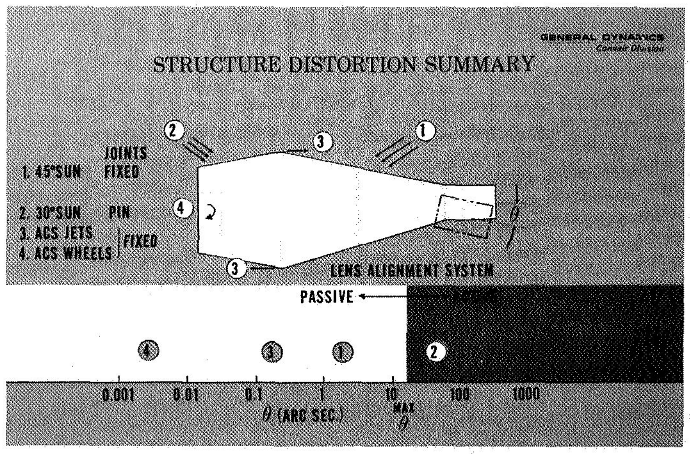



Figure 4-8

structure. The x-ray telescope, like most orbital vehicles, is not subjected to any high post-launch loading during its primary mission with the exception of docking loads, which do not impose critical design constraints on the overall structure. The most serious environmental effect on the telescope structure is distortion caused by uneven solar heating due to unsymmetrical sun orientations. See Figure 4-8. The initial telescope structure utilized beryllium for maximum stiffness and weight. Titanium was used when the stiffness, weight, and astronaut impact effects were analyzed. Although titanium was used on the final configuration it seems likely that another complete iteration of vehicle inertias, stiffness, and thermal distortions may well show that one of the low expansion iron-nickel alloys, perhaps Invar, would be the optimum choice. The reduction in expansion coefficient which these alloys provide will probably eliminate the requirement for the active lens alignment system proposed here. Detailed examination of the effects of a magnetic basic structure on the systems and vehicle torques must be undertaken before this choice can be made safely. The higher density iron-nickel alloys will result in a 240 lb weight penalty over titanium, which is not significant.

Truss Extension System. The forward lens truss assembly, secured inside of the primary truss during launch, is extended to the deployed position by an electromechanical cable run-around system. Forty-inch-long guide sliders on the lens truss ride on four parallel rails attached to the primary truss. The critical alignment between truss assemblies is accomplished by the lens alignment system simplifying the design of the track-slider system. See Figure 4-9.

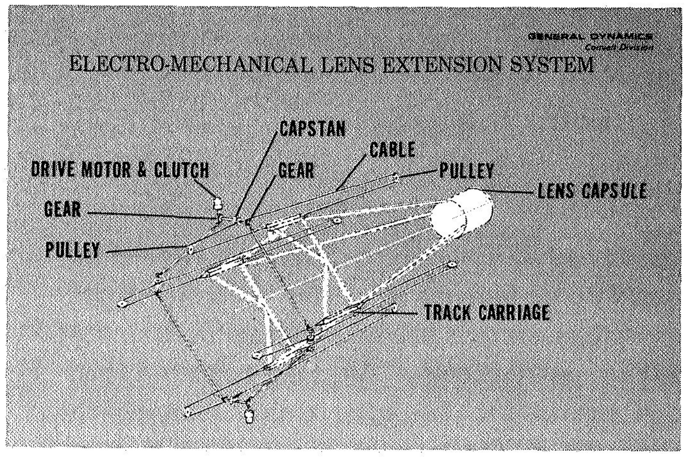



Figure 4-9

The extension system primary activation power is provided by two 1/4 hp electric motors which, through reducing gear boxes and synchronizing torque shafts, rotate four cable capstans. The cable system is pretensioned and reversible; if a failure occurs, or appears evident to the CSM crew controlling deployment, the operation may be stopped or reversed until the malfunction is corrected. The drive motors are redundant and in the event that both fail, the gear boxes have provisions for EVA portable drill motor plug-in to deploy the lens. At the fully extended position the two trusses are rigidly locked together at eight places and the four forward cable guide sheaves are shifted aft to relieve the cable tension. The cable system reliability approaches 100% with astronaut backup. The forward lens shade is automatically extended by cables during the final truss extension.

Lens Group Assembly. The x-ray telescope lens group assembly is separable from the truss assembly at one of the gimbal axes, and consists of the following subassemblies (see Figure 4-10):

a. Lens Shroud Assembly

b. Sun Filter Assembly

c. Gimbal Ring Assembly

d. Sun Shade Assembly

e. Reflecting Lens Assembly

f. Optical Telescope

g. Alignment System Reflector Package

Lens Shroud Assembly. The lens shroud assembly consists of a traveling structural support ring approximately 50 inches in diameter and 18 inches long. The ring contains trunnion points for the outer Y axis lens gimbal, holddown points for the lens launch tiedown, jack screws for adjusting the focal length, and supports for the sun shade and extension tubes.

A sun filter is mounted on the forward end of the ring to shield the lens and imaging equipment from the UV radiation during solar observations. The filter consists of four annular segments of 1/4 mil beryllium sheet spring-loaded to fail open. The filter closing mechanism is a small electric motor, cable driven; pyrotechnic cable cutters would be used to open the filter if the drive motor or gear box malfunctioned.

LENS GROUP ASSEMBLY

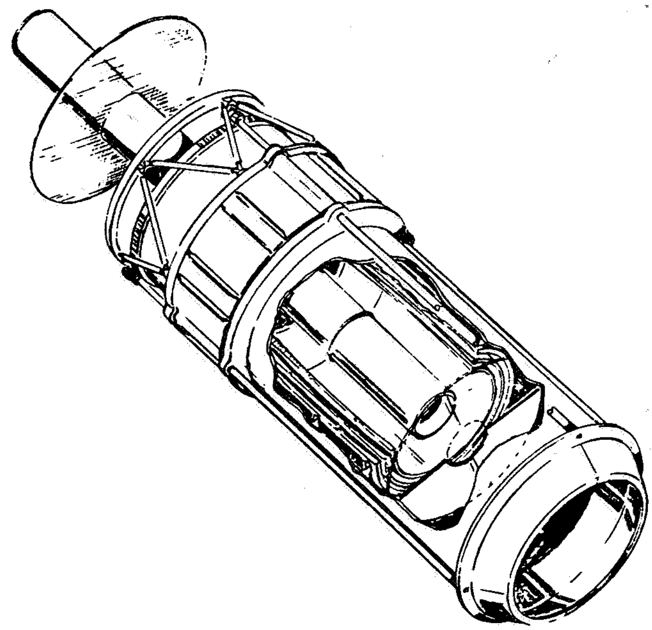


Figure 4-10

Gimbal Ring Assembly. The gimbal ring assembly is a part of the lens alignment or optical axis adjustment system. The gimbal assembly is shown on Figure 4-11.

The inner gimbal assembly provides X axis trunnions for the mirror assembly. Gimbal power for both axes is provided by stepping motors in conjunction with negligible backlash reduction gearing like the harmonic drives. High accuracy encoders sense the lens rotation angles and transmit the data to the navigation system. Changes in lens rotation constitute a shift in the vehicle optical axis, with respect to the star trackers and inertia wheels. The inner gimbal motors are extremely difficult to replace by EVA and will probably be required to operate for the entire x-ray telescope mission lifetime.

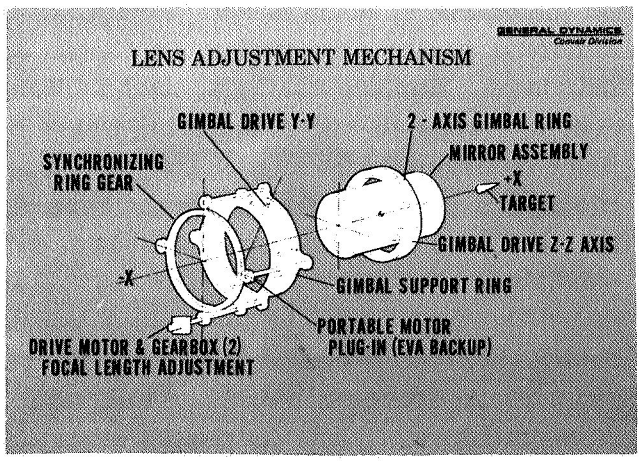



Figure 4-11

Lens Sun Shade Assembly. The sun shade assembly prevents off-axis solar radiation from striking the lens. The assembly is a tube of woven, impregnated fabric, collapsed during launch and extended by the force of the lens truss extension. The fabric is supported by a forward metallic ring attached to four tubes which are in turn supported by the lens shroud ring. The geometry shown is sufficient, with additional internal baffle rings, to shield the lens from direct solar heating when the telescope is pointed greater than 40° from the sun. A similar shade behind the lens does not require collapsing and can be attached to the lens truss. The forward shade can be designed for shielding as close as 20° to the sun. It is anticipated that sufficient stellar sources will exist to permit considerable selectivity in viewing angles with respect to the sun.

Reflecting Lens Assembly. The principal structural element in the x-ray telescope is the reflecting mirror system. The grazing incidence mirrors are formed of parabolic-hyperbolic surfaces. The effective aperture of grazing incidence optics is small compared to overall surface area because of the necessity to reflect the incoming x-rays at very small angles, below 1° in the shorter wave lengths 2 to 10 Å. Increased aperture area is best achieved by multiple confocal mirror segments; the multiple design also tends to balance the reflection coefficients throughout a wider range as each mirror has a different efficiency due to the change in grazing angle. See Figure 4-12.

MIRROR ASSEMBLY

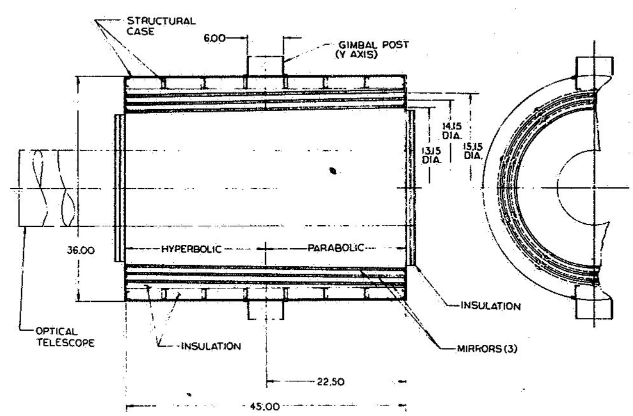



Figure 4-12

The theory and design of grazing incidence lenses is covered in detail in Volume IV.

American Science and Engineering, Inc. provided Convair with the general mirror surface tolerances required to meet the scientific community's measurement objectives.

A parametric study was performed to evaluate the effect of the 20- to 40-inch lens size range on the total vehicle. This study was based on preliminary costs, which increased as detail was developed; however, the percentage change in the various areas remained roughly constant and therefore the results are believed to yield significant trends.

Simplified equations were developed for lens sizing, weight, aperture area, and optical (surface) area. These values were used with specific costing assumptions. The results of the analysis are shown in Figure 4-13. The curves indicate that the unit hardware cost per unit area of aperture decreases with lens sizes at least up to 40 inches. This

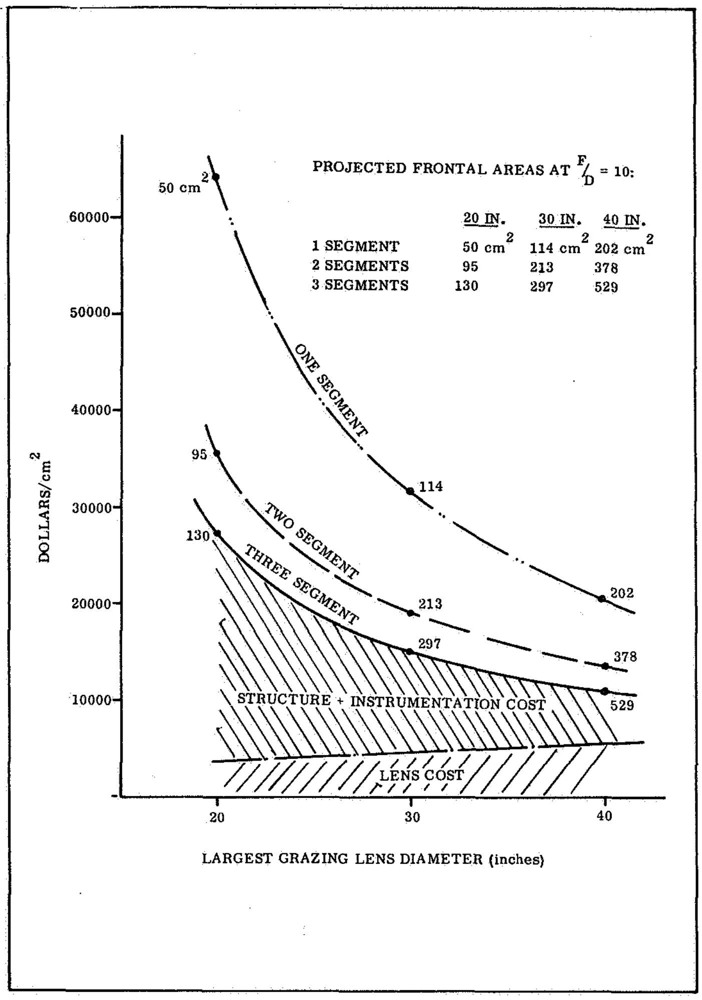



Figure 4-13

is expected because of the high cost of instrumentation equipment and vehicle subsystems which are nearly constant for lens diameters between 20 and 40 inches. The increasing cost effectiveness would be expected to reduce up to lenses of 50 inches. Beyond this size the cost-aperture slope is uncertain as new structure concepts are required because of packaging limitations.

Mirror Materials. A cursory evaluation of lens materials has been conducted primarily from a feasibility aspect. No attempt can be made to optimize mirror materials until a mission is defined specifying the primary spectrum of interest, vehicle to sun orientations, etc. Current x-ray mirrors are being manufactured of fused silica, beryllium, 440 stainless steel, electroformed nickel, and aluminum. Surface coatings, such as Kanogen, are used to add a material which has more desirable reflection characteristics or is easier to polish or figure than the base material. The mirror material selection is quite sensitive to the thermal environment and detailed thermal transient analyses of various designs will be necessary to calculate the exact temperature excursions and distortions of the mirrors. If the increased weight can be neglected, which appears true for the sizes up to 40 inches, the low coefficient glasses are a probable choice because of better thermal stability.

Image Capsule Assembly. The image capsule is the prime reference base for the orbiting vehicle. It contains the mission scientific receiving equipment, the navigating equipment, power conditioning, command/control and telemetering equipment, and docking provisions. The capsule is temperature controlled, but is not pressurized. See Figure 4-14.

IMAGE CAPSULE

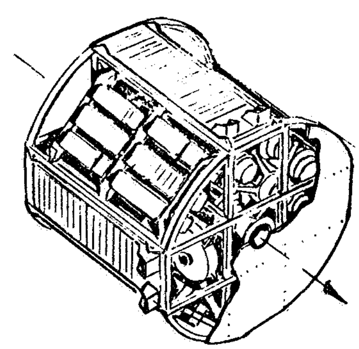


Figure 4-14

Geometrical Arrangement. The basic capsule diameter is 54 inches and the length is 48 inches. Insulation is added outboard of these basic dimensions. The structural arrangement consists of cross and vertical beams, walls, and tension straps. Double cruciform beams 15 inches apart form bulkheads at the forward and aft ends of the module. Post members connect the forward and aft bulkheads at three of the four apexes formed by the double cruciform. The fourth apex is the pivot axis for the x-ray receiving equipment turret. The primary construction material is 2024 aluminum alloy. All internal equipment and walls are accessible to an astronaut through the access doors. Most equipment is immediately inside the access doors on swing-out racks.

The capsule contains four access bays. Bay I contains the x-ray receiving equipment mounted on a turret which has seven stations, two imaging, two polarimetry, one crystal spectrometer, and two empty stations for unassigned equipment. The turret rotates each station to the focal plane of the primary x-ray mirror. See Figure 4-15.

The image capsule access doors are louvered thermal control doors with built-in foot constraints to permit the EVA astronaut to secure himself to the door and open it by portable electric drill motor. The astronaut is then securely anchored to a good work site for access into the image capsule.

<u>Docking Assembly.</u> The telescope satellite is equipped with a LEM docking cone supported on shock mounts for docking with an Apollo CSM. Docking is required for the initial extraction of the telescope from the SLA and post deployment orbital docking for servicing and/or repair.

<u>Solar Panel/ACS Assembly.</u> Four A-frame support booms are hinged from the main cross members of the primary truss in the same plane as the SLA attachment points. The frames support four solar panels and four attitude control modules consisting of three 1/2 lb $H_2O_2$ thrusters. The assemblies are deployed prior to truss extension, and once deployed provide vehicle stabilization. The boom deployment mechanisms are reversible permitting the astronauts to replace ACS modules and solar panels while firmly anchored to the primary truss structure. The ACS modules are designed so that new packages can be attached on the outboard end of the expended module only with a quick release mechanical fitting and an electrical connection. The solar cell panels are rigid cells on flexible substrate to permit them to be stowed, and transported to the worksite in compact rolls. See Figure 4-16.

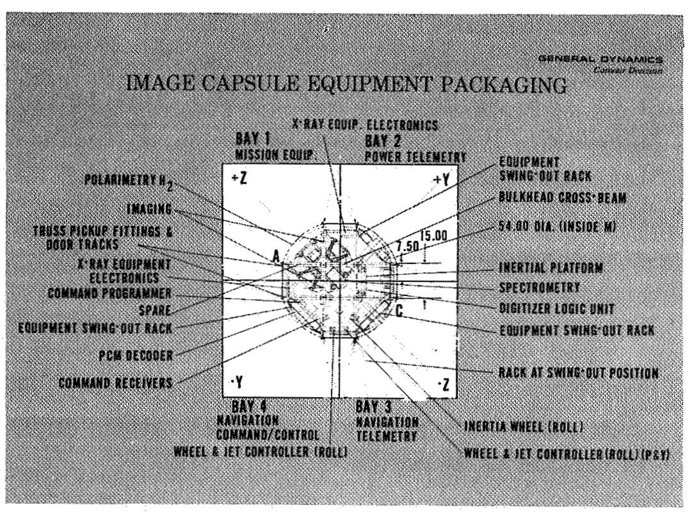



Figure 4-15

SOLAR PANEL/ACS ASSEMBLY

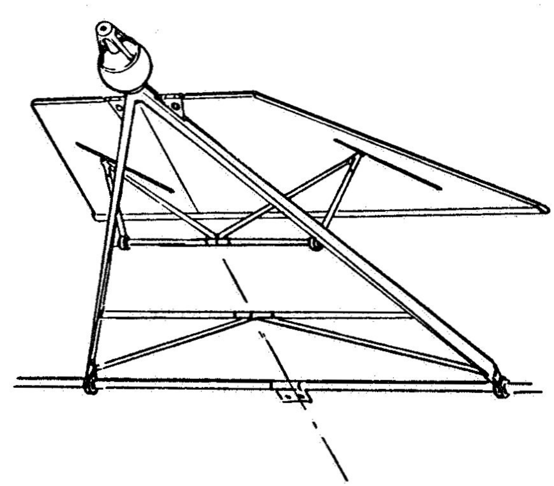


Figure 4-16

Launch Support and Separation Systems. In addition to the four primary SLA reaction points, the x-ray telescope design includes provisions to support the following assemblies during the boost phase: forward truss, ACS/solar panels, reflective mirror assembly, and the image capsule equipment turret. All of these systems are designed for remote release with EVA backup except that the turret restraints are removed by EVA.

4.4.2 SUBSYSTEM DESIGN. The following vehicle subsystems were synthesized for the x-ray telescope in order to develop a realistic total vehicle predesign regarding weight, volume, and power requirements, in addition to investigating subsystem feasibility: electrical power and distribution, command, navigation, data, telemetry and computerized attitude control. The mission-oriented telescope instrumentation and optical drift tracking subsystems were conceived through consultation with American Science and Engineering, Inc.

The lens alignment subsystem was designed after structural analysis indicated that the thermally induced distortions of the structure would, at times, exceed the maximum allowable limits of 8 to 15 arc seconds.

Electrical Power System. The electrical power subsystem consists of an omnidirectional, fixed solar panel array, batteries, and conditioning and distribution equipment.

The average power to supply the various systems will be 260 watts with a peak power of 450 watts. Solar cell power is supplied for an average of 65% of each orbit. The solar cell system consists of four 35-square-foot panels designed for projected performance of 13.5 watts/ft$^2$. These values are slightly more conservative than industry projections for the 1972 time period.

The required battery capacity is 580 watt hours based on a 30% depth of drain; three such batteries are carried to enhance system reliability. The Mallory Company Amp-Gate Charging System was assumed for charging control. The total weight of the electrical power system is 304 lb and the internal volume requirements are 4960 cubic inches. The externally mounted solar cells occupy 7-1/2 cubic feet.

Command System. The command system is designed to respond to approximately 100 remote command functions in the operation of the telescope and associated equipment. The commands will include deployment, vehicle pointing, control of sun shield, gratings, instrument turret, and selection and modes of operation of data storage and multiplexing equipment.

The command system receives digital data from ground stations, certifies it, determines the system and function to which it is addressed, and routes it to the function. The system consists of two command receivers, a signal selector, a detector-decoder, interface and buffer storage, relay drivers, and a relay assembly.

The programmed codes will operate relays, reload or up-date computer instructions, update timing generator with time works, reset relay groups, and test the system. The received command work will be checked and returned on telemetry link for verification. If errors occur, retransmission will be requested. If verification is made, an execute command is transmitted from the ground. A tone decoder will also be included for backup and emergency operation.

The command system weighs 11.5 lb and requires 248 cubic inches of space internally. The command system maximum power consumption is 15 watts.

Navigation System. The x-ray telescope navigation system utilizes ground-based sensor and computer facilities as much as possible to improve orbital system reliability since the information to be transmitted is small, and continuous ground contact is not required. The ground-based functions will include orbit determination, orbit integration, source programming, star tracker pointing and control of drag velocity loss.

The vehicle operates with two star trackers and a solar sensor; a third backup star tracker is installed and kept shielded from the space environment until needed. The solar sensor aids initial orientation and will maintain coarse vehicle stabilization for docking in the event that two star trackers fail.

The vehicle computer is similar to the IBM $4\pi$ general purpose, stored program, binary operated airborne computer.

A simple inertial reference unit utilizing three displacement gyros is included for dark side stabilization or if the reference stars become occulted.

The navigation system weight and volume estimates are 191 lb and 5100 cubic inches; the system maximum power is 240 watts.

Data, TLM and Communication System. The data handling system designed for the x-ray telescope is a conventional PCM telemetry system consisting basically of a signal conditioner, analog-to-digital converter, input selector, digital formatter, PCM multiplexer, time generator, tape recorder/converter, pre-mod processor, transmitter and two antennas, and associated multipliers, filters and selectors.

The tape recorder capacity required is approximately 8 million bits, and the maximum transmission rates to ground will be approximately 260 kbps. A single machine can handle this load with a second as backup. The total system weight and volume requirements are 44 lb and 1180 cubic inches. The maximum system electrical power is 90 watts.

Optical Drift Error System. The drift error system permits the x-ray telescope to effectively lockon optical sources in the general vicinity of the x-ray source being observed. The x-ray sources of interest (less than $10^{-4}$ photons/cm$^2$/sec) do not have sufficient flux intensities to permit vehicle lockon to the source being observed. It is necessary, therefore, to provide a guidance system which can lock on to reference stars of adequate intensity. The vehicle star tracker would limit the long term image resolution to at least 10 arc seconds, which would be increased by the vehicle attitude control system. The sensor for the drift error system consists of an optical telescope which is coaligned with the primary x-ray mirror assembly. The optical telescope has a field of view of 2 degrees assuring that one or more stars of the tenth apparent magnitude will always be within the field of view. An image intensifier and vidicon record the moving reference star images; the star drift is translated into error signals which are transmitted to the x-ray image motion compensator. The image motion compensator will require a maximum travel of 1 arc minute in the Y and Z planes, the vehicle pointing accuracy. The drift error system should permit extremely accurate imaging without requiring the vehicle to maintain precision attitude control. The drift system will not compensate for vehicle distortions or jitter between the lens assembly and the focal plane. The system weight and volume are 244 lb and 14,200 cubic inches. The system maximum power is 24 watts.

Telescope Instrumentation. The mission subsystem required to satisfy the scientific flight objectives include imaging, spectromatic, and polarimetric equipment. The actual equipment for a flight x-ray telescope facility would be the responsibility of the principal investigators and is not presently defined.

To achieve a realistic vehicle predesign it is necessary to at least synthesize the critical equipment parameters of weight, volume, and electrical power required as well as define any unusual support requirements such as temperature control.

Electronic imaging techniques are preferred to provide nearly real time access to the observation and minimize the film stowage-shielding-retrieval problems. The scientific equipment is mounted on a turret which rotates to locate each of the seven equipment stations in the focal plane. Two of the stations contain imaging packages, not necessarily redundant. The basic imaging package consists of an image intensifier, image motion compensator (a portion of the drift error compensator system) and an electronic imaging tube with long integration time capability. The imaging equipment is used in conjunction with a slitless spectrometer grating, which is positioned immediately behind the grazing incidence mirrors. This system provides spectrographic resolution on the order of $\lambda/\Delta\lambda = 100$.

The third instrument included is a focusing crystal spectrometer for precision ($\lambda/\Delta\lambda = 1000$) spectrometric measurements. Two polarimeter packages are included with the equipment parameters based on the cryogenic hydrogen design. Two equipment stations are available for scientific focal plane instrumentation.

As mentioned previously, the baseline design allowances shown are for the specific equipment assumptions covered and should not be regarded as limiting values. Considerable latitude exists in the vehicle design to incorporate larger or heavier scientific packages of any geometry. The image capsule diameter and shape can be changed. It might even be effective to increase the capsule length at the expense of reduced focal length to accommodate specific instrumentation. The baseline research equipment block diagram is shown on Figure 4-17.

RESEARCH EQUIPMENT BLOCK DIAGRAM

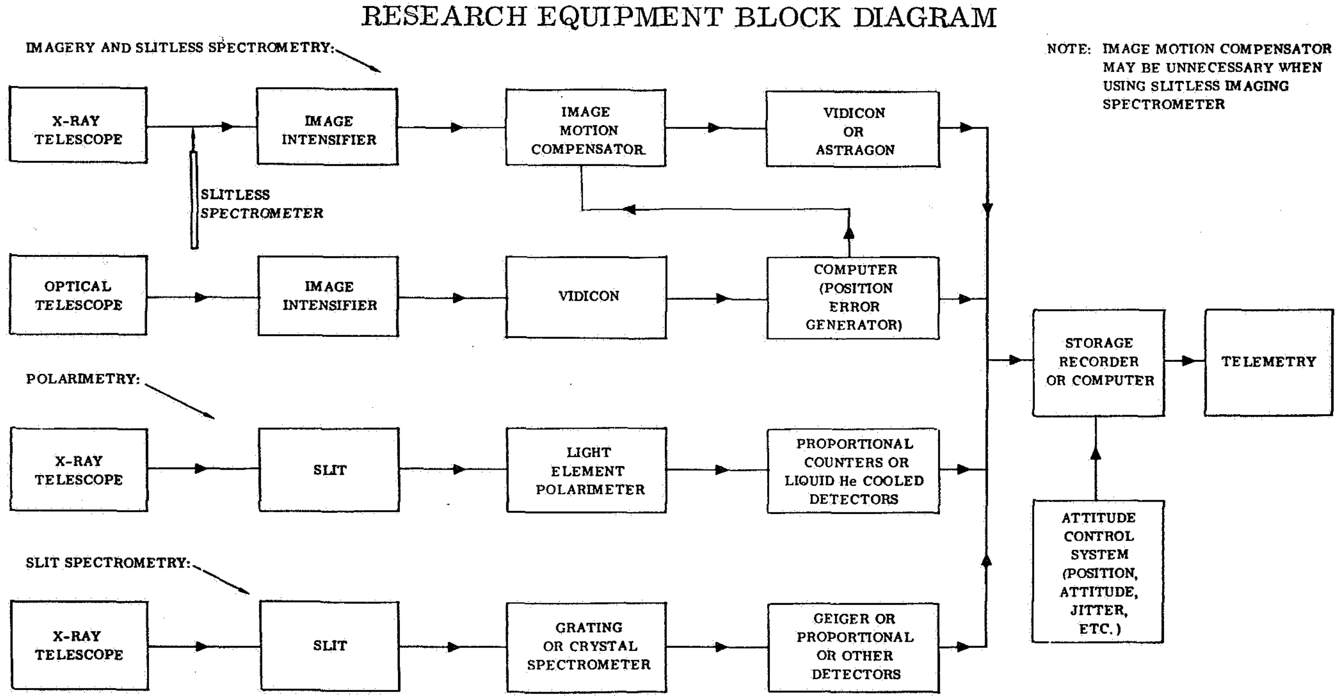



Figure 4-17

Lens Alignment. The lens alignment system compensates for the distortions of the primary truss structure. The system consists of a gallium arsenide laser and sensor package mounted on the image capsule and a reflecting mirror assembly attached to the lens assembly.

A laser beam transmitted parallel to the optical axis is reflected by the plane and corner cube mirrors at the lens. The reflected beam impinges on the sensors and the beam location is registered. The plane mirror senses lens angle changes while the corner cube reflector senses lens displacement perpendicular to the optical axis.

As structural distortions occur the resulting changes to the optical axis are calculated and corrected for by rotating the two axis-gimballed lenses. The optical axis changes are programmed into the navigation system to modify the star tracker reference axis. The system weight and volume are 12 lb and 186 cubic inches. Maximum power is 9 watts.

Attitude Control System. The telescope control system consists of three independent attitude orientation systems and a drag correction system. The orientation systems are the 12 hydrogen peroxide jets (0.5 lb thrust), the precision 3 axis inertial wheel, and the emergency cold gas docking stabilization system.

The ACS is sized to maintain the required pointing accuracy of $\pm$ 0.5 arc minutes and drift rates of 1.0 arc sec/sec for up to 90 minutes (one orbit). The lockon time can be extended to six hours with a total vehicle weight penalty of approximately 400 lb. Hydrogen peroxide was selected as the propellant on the basis of a two year mission without resupply. See Figure 4-18.

The nominal 30-inch x-ray telescope configuration has 0.0403 and 0.00034 ft-lb of

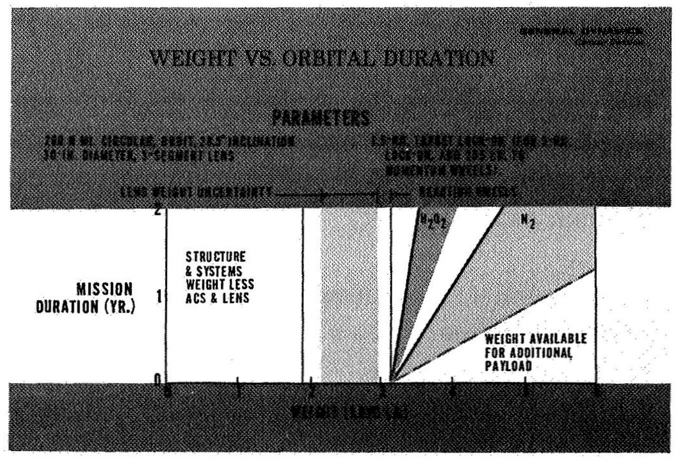



Figure 4-18

gravity gradient and aerodynamic drag torques respectively. The inertia wheel system capacity is 10 ft-lb-sec in roll and 84 ft-lb-sec in pitch and yaw.

The four 3-jet modules each have individual propellant tanks; resupply of these modules is performed by EVA; replacement units are attached to the expended unit. The precision inertia wheels are located in the image capsule assembly in close proximity to the star trackers to minimize reference coordinate errors resulting from structural distortion. The backup docking system is provided to permit remote controlled astronaut stabilization of the telescope for docking. This emergency system is only required if both normal stabilization systems are malfunctioning or the control electronics fail. The reserve system has low propellant requirements permitting the use of the highly reliable nitrogen gas thrusters. The total ACS, including propellant for two years, weighs 575 lb, occupies 5,400 cubic inches, and requires 33 watts maximum power.

4.4.3 THERMODYNAMICS. Influence of the orbital thermal environment was investigated for the major assemblies of the x-ray telescope: structure, lens, and image capsule. Temperature transients, resulting from passage through the earth's shadow, and spacecraft reorientation affect the design of each of these assemblies; non-uniform heating primarily affects the structure and lens designs. Minimum exposure is about 62%, while a maximum of about 81% sun can occur at least twice a year.

Maximum circumferential temperature gradients were calculated for the tubular structure members. These gradients contribute to bending distortion while a change in absolute temperature contributes to distortion by linear growth. Gradients are minimized with low solar absorptance coatings, reduced diameter, and increased thermal conductivity and wall thickness. Warmup time upon emerging from the earth's shadow was computed to be about 7.4 minutes while cooldown time was about 24 minutes.

The lens assembly is sensitive to thermal distortion. Segment radius growth, axial bowing, circularity, and concentricity are affected. Initial calculations indicate that temperature gradients which affect circularity and concentricity can be controlled by using an efficient external insulator such as multi-layered aluminized mylar covered with a metallic shield having good conductivity and a low $\alpha_S/\epsilon$ coating, such as white paint. Internal components should be thermally coupled and made from high conductivity material. It will be necessary to perform a detailed transient temperature analysis to determine whether insulation is adequate to minimize the effects of orbital temperature excursions on radius growth.

Calculations of image capsule temperatures using average orbital heating values established the requirement for active thermal control to maintain the temperature of 70° F $\pm$ 5°. The heating excursions result from sun exposure variation. This can be accomplished with a louver control system covering about 30% of the capsule surface.

4.4.4 CREW SYSTEMS. The role of man in the x-ray telescope is two-fold: he is the subject of EVA experiments and at the same time performs operational maintenance and repair to increase mission effectiveness and extend mission lifetime.

The integration and evaluation of man's role in the orbiting x-ray telescope has been guided by frequent coordination meetings with MSC personnel, and NAA/S&ID EVA study team. The NASA/MSC "Baseline Astronaut for 1968-72" document was utilized in establishing EVA astronaut and equipment capabilities. The EVA philosophy for the proposed design assumes that the CSM is docked to the telescope for all EVA, a 50-foot tether is used, life support is provided by PLSS, and locomotion is accomplished by hand using basic truss or hand rails; a hand gun would be carried for alternate locomotion and rescue.

The mechanical design must be such that equipment is replaced with gross clamping actions, no pneumatic or hydraulic lines are disassembled, and automation is used wherever practical with manual override capability built in.

A summary of the mechanical systems in Figure 4-19 shows the degree of planned man participation in the system operation. The scheduled crew activities for the deployment

GENERAL DYNAMICS
Convair Division

MECHANICAL SYSTEMS

| | | FUNCTION PRIMARY | FUNCTION SECONDARY |
|---|---|---|---|
| SINGLE CYCLE SEPARATION | SLA/TRUSS | REMOTE | — |
| | SOLAR/ACS/TRUSS | REMOTE | EVA |
| | FWD. TRUSS TO AFT TRUSS | REMOTE | EVA |
| | LENS/FWD./TRUSS | REMOTE | EVA |
| | TURRET/IMAGE CAPSULE | EVA | — |
| EXTENSION | FWD. TRUSS | REMOTE | EVA |
| | FOCAL LENGTH | REMOTE | EVA |
| | LENS SHROUD | REMOTE | EVA |
| TWO OR MORE CYCLES | SOLAR PANEL/ACS BOOM | REMOTE | EVA |
| | DOCKING ASSEMBLY | REMOTE | — |
| | IMAGE CAPSULE DOORS | EVA | — |
| | SOLAR FILTER | REMOTE | EVA |
| MULTIPLE OR CONTINUOUS CYCLING | LENS GIMBALS | REMOTE | EVA* |
| | TURRET | REMOTE | EVA* |
| | THERMAL LOUVERS | REMOTE | EVA* |
| | SPECTROMETER GRATING | REMOTE | EVA* |

*EVA USED TO REPLACE/REPAIR, IF POSSIBLE; OTHERWISE CREW SETS BEST POSITION

Figure 4-19

checkout, resupply, and refurbishment phases are shown on Figure 4-20. Some of the specific EVA design features which are required to permit man to effectively repair and resupply the vehicle are the roll up solar panels, add-on ACS modules, access door foot restraints, and provisions on all the electrically powered mechanical systems for a portable hand held torque motor. The roll up solar cells simplify transfer of new panels from the resupply vehicle to the worksite.

Task analysis forms were prepared to identify EVA, equipment, and design requirements. See Figures 4-21 and 4-22. The detailed task analysis was generally limited to normal operations in the resupply phase because of study priorities. A separate timeline analysis was prepared for the initial deployment and operational checkout phases. The initial deployment time (to operational status) requires 44 hours of which 2 hours and 18 minutes is EVA. The resupply mission replaces the ACS, solar panels, and three scientific packages and requires 4 days, 18 hours on station and 17 hours and 32 minutes of EVA. The extent of the refurbishment tasks cannot be estimated without detailed component operating lifetimes; therefore, no time line analysis was attempted. A conservative estimate of refurbishment would be at least double the resupply task.

4.4.5 DYNAMICS. The dynamic behavior of the 30- and 40-inch x-ray telescope structures is satisfactory. The lens and truss assembly deflections during the observation period are well within the specified tolerances.

The practice of using very conservative analytical techniques to bound magnitudes of dynamic response has been used extensively in this study. As the design details become more firmly established, it will be appropriate to use more exact analysis tools.

The dynamic analysis of initial docking has shown that the docking truss requires

GENERAL DYNAMICS
Convair Division

SCHEDULED CREW ACTIVITIES

| IVA | EVA |
|---|---|
| **DEPLOYMENT** | **POST DEPLOYMENT** |
| OBSERVE IMAGE CAPSULE STATUS PANEL & CM DISPLAY PANEL | CHECK ACS/PANEL FRAME & TRUSS LOCK |
| COMMAND ACS/SOLAR PANEL DEPLOY | REMOVE TURRET LAUNCH RESTRAINTS |
| COMMAND LENS TRUSS EXTENSION | OPERATE TURRET - MANUAL & AUTO. |
| COMMAND LENS LAUNCH RING RELEASE | CYCLE SPECTROMETER GRATING & SUNSHADE |
| ALIGNMENT/ADJUSTMENT | ALIGNMENT/ADJUSTMENT |
| MONITOR SYSTEMS | OBSERVE LENS GIMBALING |
| | ADJUST LENS ALIGNMENT SYSTEM |
| **OPERATIONAL CHECKOUT** | **OPERATIONAL CHECKOUT** |
| MONITOR GROUND SYSTEM CHECKS, VERIFY | |
| COMPARE TELESCOPE POINTING TO CM SYSTEM | |
| **RESUPPLY** | **RESUPPLY** |
| MONITOR SYSTEM | ACS/POWER (PANELS & BATTERIES) |
| | MISSION IMAGING EQUIPMENT |
| **REFURBISHMENT** | **REFURBISHMENT** |
| MONITOR SYSTEMS | MISSION IMAGING EQUIPMENT |
| | IMAGE CAPSULE OPER. SYSTEMS |
| | LENS CAPSULE OPER. SYSTEMS |
| | ACS/POWER (PANELS & BATTERIES) |

Figure 4-20

GENERAL DYNAMICS
Convair Division

SAMPLE ORBITAL TASK ANALYSIS (ABBREVIATED)
9.0 Operational Checkout, Mission Equipment

NORMAL OPERATION

| GROSS FUNCTION | SYSTEM OR COMPONENT FUNCTION | EVENT TIME | ELAPSED TIME | DISPLAY INDICATION | CREW ACTION OR PARTICIPATION |
|---|---|---|---|---|---|
| 9.1 LENS/IMAGE RECEIVER ALIGNMENT | 9.1.1 TELEM SYSTEM ON | | | CM & GROUND DISPLAY PANELS | ONE ASTRO. EV AT IMAGE CAP. OPER. CENTRAL SWITCH |
| | 9.1.2 LENS GIMBAL SYSTEM ON | | | CM & GROUND DISPLAY PANELS | EV ASTRO. AT IMAGE CAP. OPER CENTRAL SWITCH |
| | 9.1.3 LASER BEAM TRANSMITTER ENERGIZED | | | CM & GROUND DISPLAY PANELS | EV ASTRO. AT IMAGE CAP. OPER. CNTL. SW. OBSERVES BEAM TARGET |

ABNORMAL OPERATION

| | EVA EQUIPMENT | POSSIBLE EVA HAZARDS | SAFETY OR EMERG. PROC. | FAILURE MODE | FAILURE INDICATION |
|---|---|---|---|---|---|
| P. | PORT. HAND PWR. TOOL, PLSS, TETHER | NORMAL | NORMAL | | |
| | SAME | NORMAL | NORMAL | | |
| | SAME | NORMAL | NORMAL | XMITTER DOES NOT GENERATE BEAM | CM & GROUND DISPLAY PANELS VERIFIED BY EV ASTRO. |

Figure 4-21

GENERAL DYNAMICS
Convair Division

SAMPLE ORBITAL TASK ANALYSIS (ABBREVIATED) (Cont.)
9.0 Operational Checkout, Mission Equipment

ABNORMAL OPERATION

| FAILURE INDICATION | CREW ACTION OR PARTICIPATION | EVA EQUIPMENT |
|---|---|---|
| DISPLAY PANELS EV ASTRO. | UNIT REPLACEMENT EV ASTRO. OBTAINS NEW TRANSMITTER FROM STOW. MANUAL TO AFT EXT. OF OPT. TELES., REMOVES UNIT, INSTALLS NEW UNIT. LASHES SPENT UNIT TO STRUCTURE. RETURNS TO IMAGE CAP. & PROCEDES WITH CHECKOUT | PLSS, TETHER, PORTABLE HAND POWER TOOL |

| POSSIBLE EV HAZARDS | SAFETY OR EMERGENCY PROC | REMARKS |
|---|---|---|
| TETHER ENTANGLEMENT-DISTANCE $\cong$ 35 FT. FROM CM | ADEQUATE RESTRAINT AT WORK SITE. TETHERS ON TRANSMITTER | LASER TRANSMITTER IS REPLACEABLE. CAPTURED FASTENERS, INTEGRAL TETHERS FOR HANDLING & LASH TO STRUC. ELEC. DISCON. |

Figure 4-22

shock attenuation additions. The necessary change in stiffness is feasible and the orbital docking dynamics will then be satisfactory by a large margin. Separation of the CSM/ telescope from the S-IVB and deployment of the telescope are not operations which pose design feasibility questions.

The dynamic response of the 30-inch diameter x-ray mirror assembly was analyzed assuming a mirror thickness of 3/16 inch, (weight estimate is based on 3/8 average). The worst case vibratory distortions of the lens are well below the microinch tolerance. The telescope structure was examined similarly for the attitude control jets and inertia wheels; the truss distortions during observation were computed to be several orders of magnitude below allowable limits. No problems with dynamic structural response are anticipated for this size telescope structure or lens.

4.4.6 MASS PROPERTIES. The total x-ray telescope weight with titanium primary structure and a 30-inch, three-segment beryllium mirror is 2700 lb. If fused silica or low coefficient glass is used for the x-ray mirror assembly, the total vehicle weight will increase to 3570 lb.

The nominal 30-inch x-ray telescope weight summary is:

| | |
|---|---|
| Structure & Mechanisms | 661 |
| Reflecting Lens Assembly (Beryllium) | 292 |
| Attitude Control System (2 year) | 575 |
| Command/Control | 11 |
| Telemetry, Data & Communication | 44 |
| Navigation | 191 |
| Electrical Power | 304 |
| Lens Alignment | 15 |
| Scientific Instrumentation | 360 |
| Contingency & Growth (10%) | 247 |
| | 2700 |

The nominal 30-inch vehicle mass moments of inertia are 2500 slug $ft^2$ and 13,800 slug $ft^2$ for roll and pitch (yaw) respectively.

4.4.7 STRESS ANALYSIS. The analysis of the truss structure was performed by digital computer using well known displacement programs.

Two truss analysis models were used, one assuming rigid joints, the other assuming pinned joints. The fixed, jointed model was loaded with unit shear and moments to determine influence coefficients for the dynamic analysis. The fixed truss distortions were also computed for the orbital docking conditions and 45° relative sun condition. The thermal analysis of the fixed joint model considers only structure distortions caused by member bowing. The effect of individual member thermal growth on the truss distortions was analyzed for the pinned joint model. This is the critical condition which results in 130 arc secs of distortion over the 30-foot focal length. The magnitude of distortion is exaggerated by the initial assumption that the truss member temperatures were stabilized at sub solar point in orbit. The effects of earth thermal and albedo radiation were neglected at the sub solar point, resulting in maximum member temperature differentials and increased distortions.

## 4.5 RDT&E

The Research Development Test & Engineering plan provides information for defining the steps to achieve a functioning orbital x-ray telescope as part of AAP. Sufficient definition is required to support NASA resource allocations between the various alternatives and identify requirements for manpower, research, development and test facilities and to define schedule interactions and budgetary planning data to achieve for AAP maximum utilization of resources. The RDT&E plan provides:

a. Work Breakdown Structure

b. Prerequisite Orbital Experiments

c. Research, Manufacturing, Test and Support Plans

d. Schedule

e. Cost Analysis

The x-ray telescope structure can be deployed vertically to simulate zero-g with an overhead clearance of 45 feet. This was not assumed to be a new facility requirement; however 600k was included in the cost analysis to provide for facility uncertainties regarding mirror fabrication.

No firm requirements for prerequisite orbital experiments have been identified for the x-ray telescope. Orbital experiments to determine realistic astronaut locomotion loads and develop portable EVA cargo handling systems are encouraged for this and many similar space structures. These experiments will logically be performed early in the AAP.

The new technology areas applicable to the x-ray telescope are:

a. Improved Solar Cell Performance

b. Multiple Nesting Designs for Large X-Ray Mirrors

c. Structure Materials with High Thermal Distortion Stability

d. Mechanism & Motors for 3 to 4 Year Space Operation

e. Long Memory High Resolution Imaging Orthicons

f. Compact Focusing Crystal Spectrometers

g. Non-Cryogenic Polarimeters with High Efficiencies

The manufacturing processes required to fabricate components of the x-ray telescope are essentially those used extensively by airframe and aerospace manufacturers. The 3 to 4 segment 30-inch assembly used in the baseline vehicle can be fabricated using current technology. Increasing the aperture by using 8 to 10 segments will demand new fabrication technology.

The test program is conducted to provide design reliability and quality assurance such that the equipment will survive the ground and space environments with no malfunctions or deteriorating effects. The testing involves development, qualification, and acceptance tests. Component, subsystem, and system level testing is performed on the subsystem articles, engineering model (or prototype), and the flight article. No unusual problems are envisioned in the test area. The structure is considerably more rigid than most space structures and does not require unusual supporting or zero-g devices during testing.

The support plan summarizes the general requirements for all activities performed on the x-ray telescope subsequent to sell off of the flight unit. Included in the activities are: personnel training, prelaunch activities, range documentation, launch site, and orbital operations. Special training is required to develop the necessary skills in the mission-oriented tasks. In addition to the detailed training program for the astronauts, some training is required for both the launch crew and orbital support crew.

The x-ray telescope spacecraft would be supplied to the contractor's facility at MSFC for MSFN network compatibility and Saturn IB fit checks. At the completion of MSFC operations the spacecraft would be shipped to KSC for launch site operations. The vehicle would be launched from Complex 34 or 37 at KSC.

The program schedule for the 3 to 4 segment 30 inch x-ray telescope (Figure 4-23) indicates a launch approximately 5 years after a Phase B definition go-ahead. The program would be extended approximately 1-1/2 years if an 8- to 10-segment mirror assembly is desired.

The cost analysis for the x-ray telescope program based on the 3 to 4 segment mirror is shown in Figure 4-24. The 8- to 10-segment mirror design would cost an additional $3.0 million.

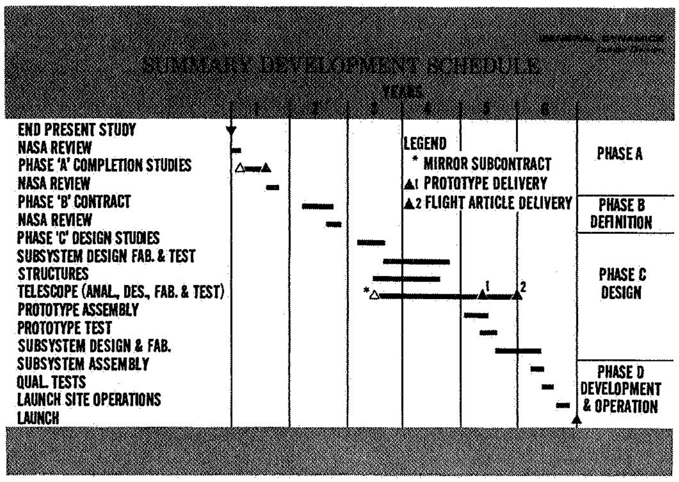



Figure 4-23

COST ANALYSIS SUMMARY

| | | |
|---|---|---|
| SYSTEM COST | | $ MILLION |
| NONRECURRING | | $30.3 |
| DEVELOPMENT | $ 28.7 | |
| GSE | 1.0 | |
| FACILITIES | 0.6 | |
| RECURRING (UNIT COST) | | 10.0 |
| TOTAL SYSTEM COST | | $40.30 |

Figure 4-24

# SECTION 5
# PARABOLIC ANTENNA

## 5.1 INTRODUCTION

The Saturn-launched experiment would deploy a large 100-ft parabolic expandable truss antenna, preferably in synchronous orbit, evaluate man's participation in a broad spectrum of IVA and EVA tasks, develop the practicality of a large deployable electromechanical system, determine the antenna's structural, dynamic, and rf performance, and, upon CSM departure, leave a large refurbishable communication system with a minimum of 2 to 5 year life operating under ground control.

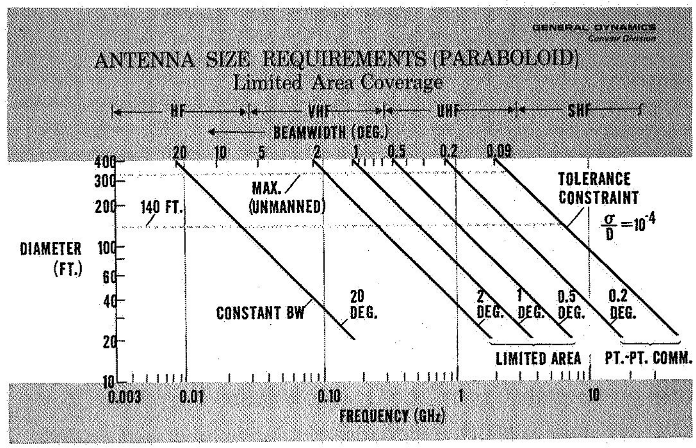



Figure 5-1

With the rapid growth of communication systems, large antennas in the 40- to 200-ft diameter range are required to provide optimum use of ever-shrinking available frequency spectrum. Large antennas, Figure 5-1, reduce the beamwidth, controlling the area of the orbital signal. From synchronous orbit a 17-deg angle subtends the hemisphere. Two degrees typically cover a 800-mile time zone in the U.S.; at high frequencies a 1/2 power beamwidth of 1/8 deg would intersect a 50-mile zone.

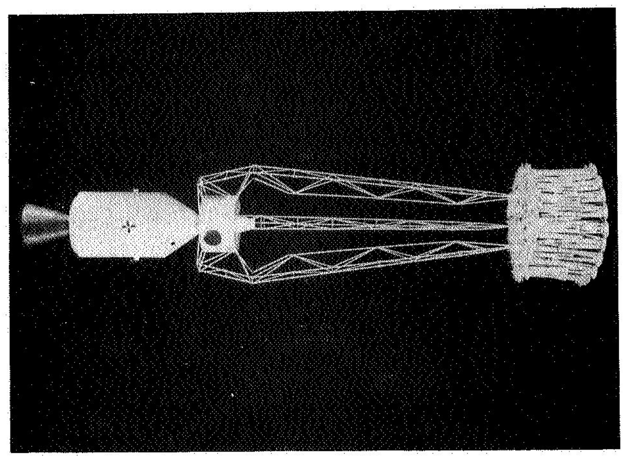


Figure 5-2

Volume in the available Saturn fairing would limit antenna size to 320-ft diameter in an unmanned launch and 140-ft diameter in the LEM adapter area with a manned CSM. A practical reflector tolerance boundary of $\sigma$(RMS)/diameter ratio of $10^{-4}$ limits the efficient useful range of the antenna and the minimum beamwidth. Peak efficient frequency application for the 100-ft diameter experiment antenna is 6 GHz, resulting in a 1/8 deg beamwidth.

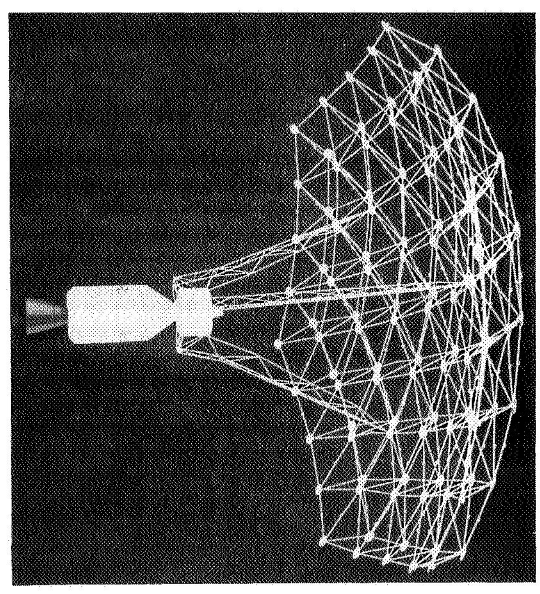


Figure 5-3

A model of the expandable truss antenna with the feed extended and the reflector in the packaged condition is shown in Figure 5-2. Figure 5-3 shows it with the reflector deployed. The Apollo CSM is docked to the pressurized feed and electronic compartment. Figure 5-4 is a photo of a second 7-ft diameter model with a mesh covering, currently in test under a related study.

## 5.2 FLIGHT OBJECTIVES

Three primary flight objectives are outlined in Figure 5-5. First, man's performance is to be evaluated using biomedical sensors as he participates in a series of orbital tasks. At a console inside of the pressurized feed/ electronics compartment docked to the CSM, the crew will direct the experiment and monitor the subsystems. Typical tasks range from reflector surface measurements, feed boresighting, and rf pattern analysis to acquisition of earth and celestial targets. The compartment is also used as an airlock for EVA tasks. EVA is required for complete inspection of the antenna, potential repairs, and, most important, adjustment of the reflector mesh for optimum performance.

The second flight objective is to demonstrate the feasibility of a large space erectable structure. Uniquely, the expandable tubular truss structure can be scaled to many different sizes and shapes. In this experiment a 100-ft diameter parabolic reflector and the expandable feed supports demonstrate its application to many future missions such as: orbital communication and surveillance, radiator and solar cell support, space station docks, magnetometer and nuclear power booms, and possibly uniting propellant tanks into an integral structure for long term missions. The tubular structure can be supported by man from hinge lockup to replacement of a damaged element. While these tasks are relatively simple using man, it would be impractical to automate them.

The scientific objective of the experiment is to analyze the full pattern measurements of the antenna in the 100 MHz to 6 GHz range.

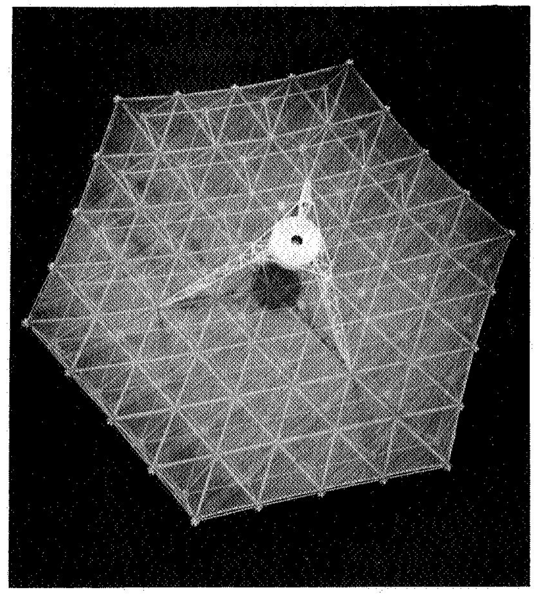
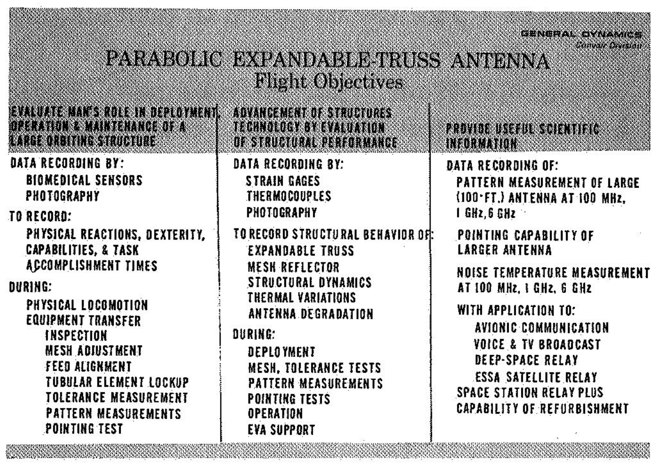


Figure 5-4



GENERAL DYNAMICS
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PARABOLIC EXPANDABLE-TRUSS ANTENNA
Flight Objectives

| EVALUATE MAN'S ROLE IN DEPLOYMENT, OPERATION & MAINTENANCE OF A LARGE ORBITING STRUCTURE | ADVANCEMENT OF STRUCTURES TECHNOLOGY BY EVALUATION OF STRUCTURAL PERFORMANCE | PROVIDE USEFUL SCIENTIFIC INFORMATION |
|---|---|---|
| DATA RECORDING BY:<br>BIOMEDICAL SENSORS<br>PHOTOGRAPHY<br>TO RECORD:<br>PHYSICAL REACTIONS, DEXTERITY, CAPABILITIES, & TASK ACCOMPLISHMENT TIMES<br>DURING:<br>PHYSICAL LOCOMOTION<br>EQUIPMENT TRANSFER<br>INSPECTION<br>MESH ADJUSTMENT<br>FEED ALIGNMENT<br>TUBULAR ELEMENT LOCKUP<br>TOLERANCE MEASUREMENT<br>PATTERN MEASUREMENTS<br>POINTING TEST | DATA RECORDING BY:<br>STRAIN GAGES<br>THERMOCOUPLES<br>PHOTOGRAPHY<br>TO RECORD STRUCTURAL BEHAVIOR OF:<br>EXPANDABLE TRUSS<br>MESH REFLECTOR<br>STRUCTURAL DYNAMICS<br>THERMAL VARIATIONS<br>ANTENNA DEGRADATION<br>DURING:<br>DEPLOYMENT<br>MESH, TOLERANCE TESTS<br>PATTERN MEASUREMENTS<br>POINTING TESTS<br>OPERATION<br>EVA SUPPORT | DATA RECORDING OF:<br>PATTERN MEASUREMENT OF LARGE (100-FT.) ANTENNA AT 100 MHz, 1 GHz, 6 GHz<br>POINTING CAPABILITY OF LARGER ANTENNA<br>NOISE TEMPERATURE MEASUREMENT AT 100 MHz, 1 GHz, 6 GHz<br>WITH APPLICATION TO:<br>AVIONIC COMMUNICATION<br>VOICE & TV BROADCAST<br>DEEP-SPACE RELAY<br>ESSA SATELLITE RELAY<br>SPACE STATION RELAY PLUS CAPABILITY OF REFURBISHMENT |

Figure 5-5

Tests of this type are not feasible on the ground since a clear sighting of 30 to 50 miles is required. Besides determining antenna efficiency, which significantly affects costly power systems, extraneous sidelobe patterns that could interfere with transmission in foreign areas are analyzed. Noise temperature measurements will be made to determine the background interference in space. Transmission and target acquisition tests will complete operational simulation of a large communication system.

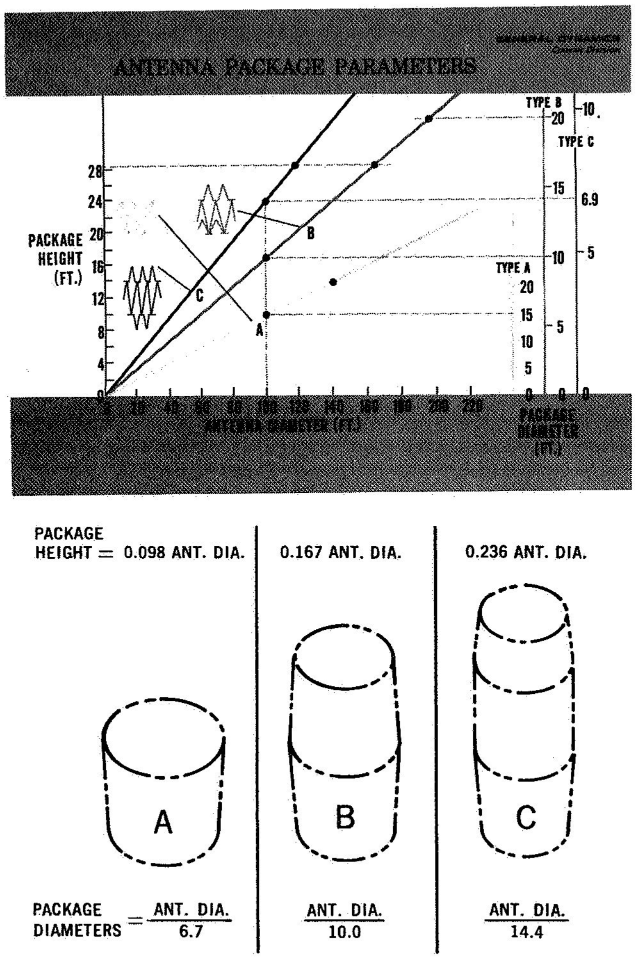


Figure 5-6

## 5.3 PACKAGING

A major factor in selection of the expandable truss is its packaging versatility. The same size antenna may be packaged in three different envelopes. Figure 5-6 illustrates the three packaging systems. "A" is a flat "coffee can" type package suitable to the LEM adapter area; in "B" the top member that hinged inward now hinges outward, reducing the package size but increasing its height; in "C" both external members hinge out, providing the tallest package but the smallest in diameter. Typically a 100-ft-diameter antenna would be packaged in a 15-ft, 10-ft, or 7-ft diameter package with the height changing inversly from 10 ft, 16.7 ft and 23.6 ft. Thus the antenna developed in this experiment can be conveniently integrated into future systems with different packaging problems. System "A" was chosen for the experiment.

## 5.4 OPERATING SEQUENCE

The packaged antenna will be supported on a pallet during transportation and during boost (Figure 5-7). A shear pin and nine redundant

OPERATION SEQUENCE

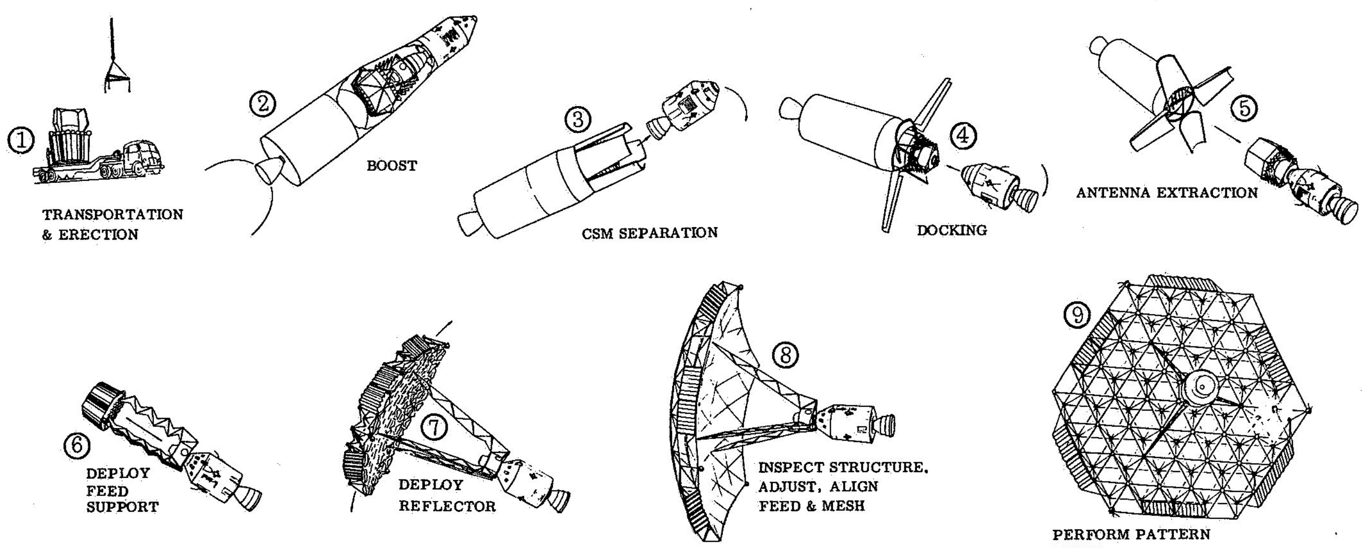


Figure 5-7

explosive bolts will position the lower spiders on the pallet, which is permanently bolted to four LEM hard points. A band holds the gear-interlocked upper spiders in position until reflector deployment. Four retractable A frames support the feed and electronic compartment and carry its boost loads directly into the LEM hard points. Boost to synchronous equatorial orbit will take approximately 7-1/2 hours.

Extraction of the antenna experiment is identical to the CSM/LEM system except for the pallet remaining with the LEM fairing. Three feed support legs are deployed at crew initiation. After checkout to assure the feed legs are structurally sound, the reflector will be deployed. IVA and EVA inspection and checkout by the crew will determine if structure and mesh are in position and if all systems are operating.

With a laser measuring unit capable of scanning the reflector in 30 seconds the contour of the mesh surface will be determined. Corrections, if necessary, will be made by the EVA astronaut to the tubular members and the mesh adjusted to optimum performance as required. Tasks and times are listed in Figure 5-8. Tolerances in the reflector will be biased by moving the feed to the optimum focal point of the best fit paraboloidal surface of the reflector. In addition to determining the surface contour, the laser will be used to boresight the feed to the mechanical axis. Pattern, noise temperature, pointing, and transmitter tests will then be made. After a final inspection the crew will secure the experiment for automatic operations and stand off to observe its performance.

The pressurized feed electronic compartment will contain most of the experiment equipment and receive environmental control from the CSM. Crew activities, planned and abnormal, are listed in Figure 5-9. Mesh adjustment for improved reflector tolerance

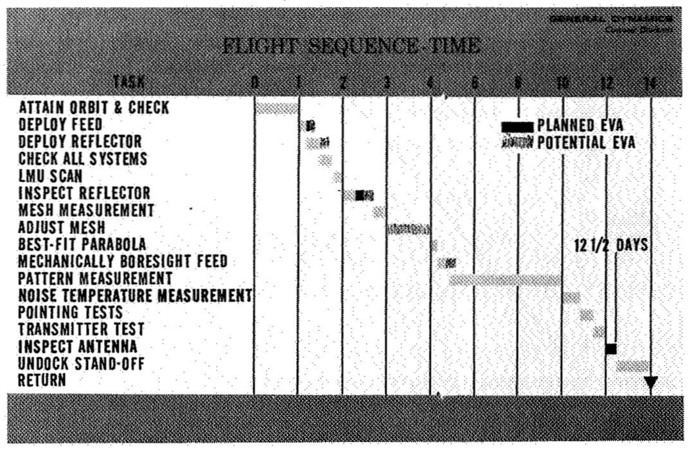


Figure 5-8

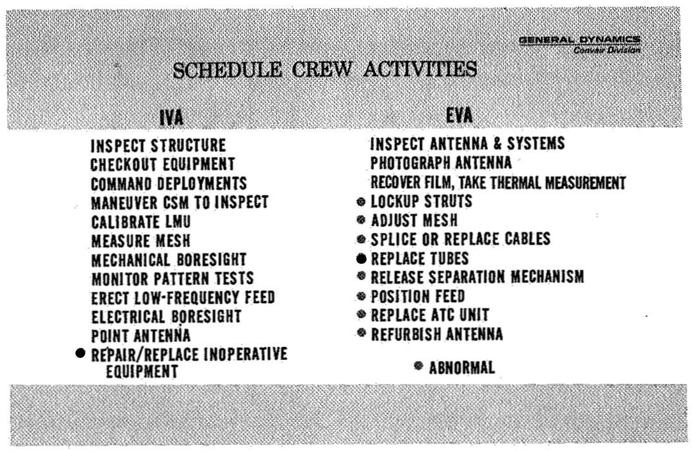


Figure 5-9

is the most likely of abnormal EVA tasks; a full day is left in the schedule for this activity. Other IVA/EVA tasks could be simulated if the experiment were completely successful: replacing a tubular element, splicing or replacing a cable, or manual adjustment of feed. This wide variety of tasks would measure man's ability to support a large electromechanical orbital structure.

## 5.5 SUBSYSTEMS

Figure 5-10 shows the orientation of the reflector mounted subsystems. The natural stiffness and strength of the expandable truss allows a large number of the systems to be mounted directly on the antenna. Dark lines indicate the power, coaxial telemetry, and command system cables. Telemetry anten-

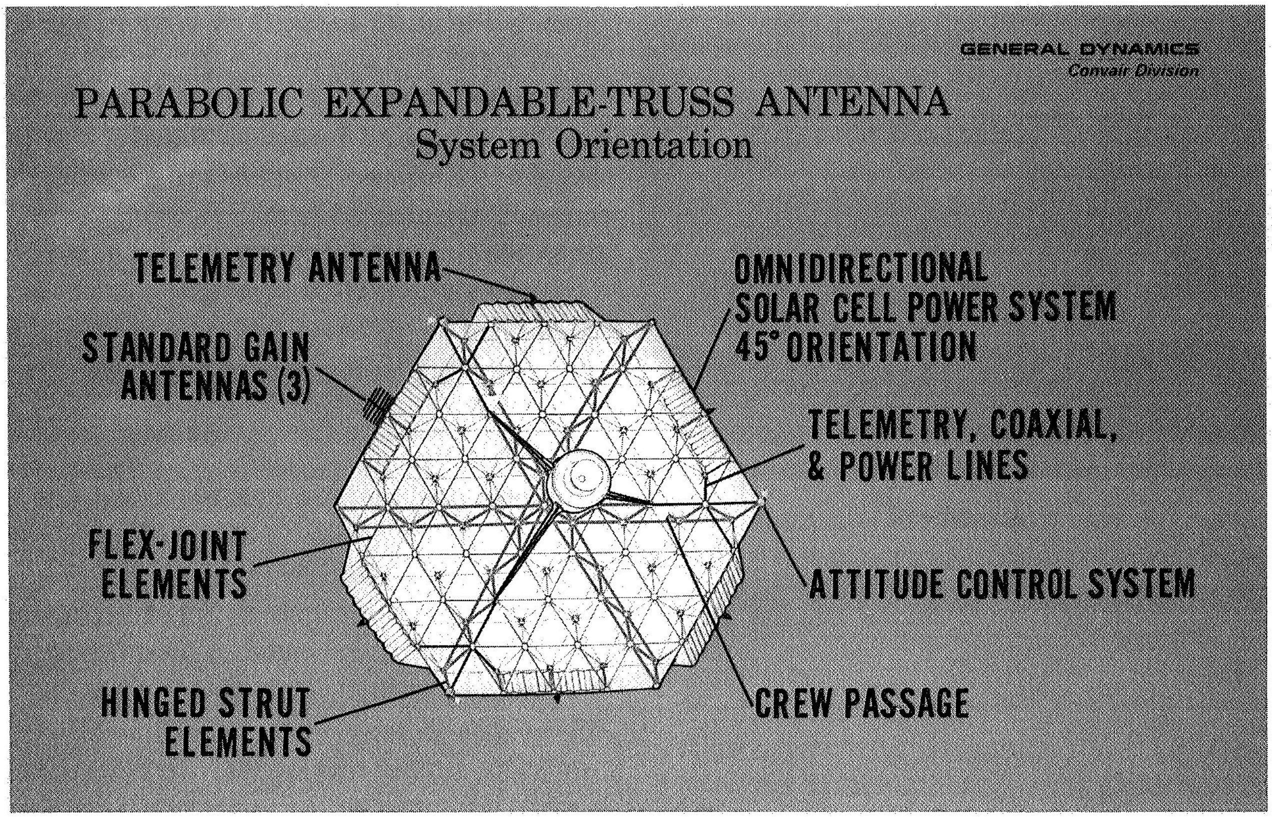


Figure 5-10

nas are mounted on the periphery for operations when the reflector shadows the CSM from the earth. A three-element cold gas expulsion system may be economically used for gross attitude control and station keeping.

A six-panel, accordian type expandable solar cell system is integrated into the outer members of the reflector to provide an omnidirectional power system. Three standard gain antennas operating at 100 MHz, 1 GHz, and 6 GHz are also mounted along the periphery. Each of the three major diagonals have hinge/lock joints with microswitch instrumentation. The joints internal to these members use the flex (mast) joint for a hinge. Crew passage to the rear side of the antenna is provided by a triangular hole in the mesh at each feed leg. Communication relay is also provided when the crew is working on the rear side of the reflector mesh. Although the mesh is 80% transparent to light, it will cut radio contact between the EVA man and the CSM.

GENERAL DYNAMICS
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ANTENNA MATERIALS EVALUATION

| MATERIAL DESIGNATION | PROPERTY | | | | | | |
|---|---|---|---|---|---|---|---|
| | $\rho$ LB./IN.$^3$ | E PSI × $10^5$ | K B/HR-FT.$^2$-°F | $\alpha$ IN./IN.-°F × $10^{-6}$ | $K/\alpha \times 10^6$ B/HR.-FT$^2$ | MATERIAL INDEX (2) | $W/W_{AL}$ (1) |
| ALUMINUM ALLOYS | | | | | | | |
| 2024-T4 | 0.100 | 10.6 | 70 | 12.6 | 5.55 | 588 | 1.000 |
| 7075-T6 | 0.101 | 10.3 | 90 | 12.9 | 6.97 | 725 | 0.775 |
| MAGNESIUM ALLOY | | | | | | | |
| AZ 31 B-H24 | 0.064 | 6.5 | 56 | 14 | 4.0 | 405 | 1.056 |
| MAG-THORIUM | | | | | | | |
| HK 31A-H24 | 0.065 | 6.5 | 61 | 14 | 4.35 | 435 | 1.411 |
| MAG-LITHIUM | | | | | | | |
| LA 141-T7 | 0.049 | 6.0 | 25 | 21.8 | 1.148 | 124 | 3.14 |
| BERYLLIUM | 0.067 | 42.0 | 87 | 6.4 | 13.6 | 8,520 | 6.79 |
| LOCKALLOY (BE 38% AL) | 0.075 | 32.0 | 123 | 9.2 | 13.4 | 5,700 | 4.02 |
| TITANIUM | | | | | | | |
| 6AL, 4V (ANNEALED) | 0.160 | 16.0 | 3.8 | 4.8 | 0.792 | 79.2 | 0.585 |

(1) MATERIAL WEIGHT INDEX WITH RESPECT TO 2024-T4 Al ALLOY FOR ASTRONAUT IMPACT
(2) MATERIAL INDEX FOR STRENGTH - TEMPERATURE=$(E/\rho)(K/\alpha)$

Figure 5-11

5.5.1 STRUCTURAL SUBSYSTEM. The expandable truss consists of hinged external and continuous diagonal tubular elements, joining spiders with mesh adjustments, and the flexible mesh system. Figure 5-11 shows that a beryllium tube would be significantly better on a weight, stiffness, thermal distortion basis. Perforated aluminum tube is proposed for use in the experimental antenna due to its availability, cost, and the

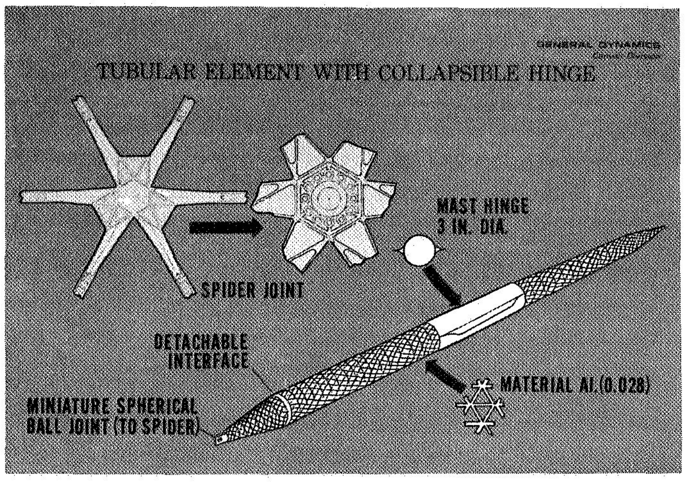


Figure 5-12

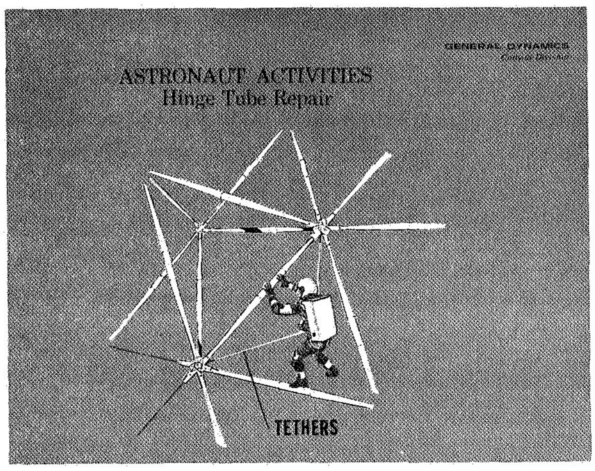


Figure 5-13

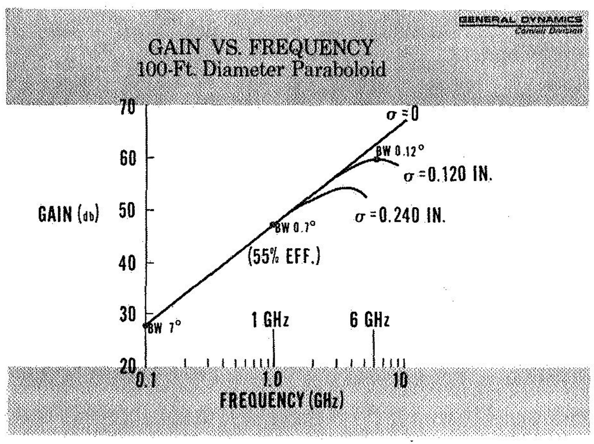


Figure 5-14

lack of a need for weight saving as the experiment is currently conceived. The present reflector weighs 0.10 lb/sq ft of aperture; with beryllium tubing weight could be reduced to 0.08 lb/sq ft or a saving of 160 lb on a 100 ft antenna. A typical tube and spider are shown in Figure 5-12. Each spider leg has a geared edge to interlock with its six neighbors. Twelve mesh adjustment screws are located in the center of the spider. Flexible hinge joints of BeCu are used on all members except the major diagonal previously discussed. EVA operation will ensure lock down and full deployment of each joint (Figure 5-13).

Reflector Mesh. The critical tolerance element of the antenna is the reflector mesh. Figure 5-14 illustrates that if the proposed surface RMS tolerance of 0.120 in. doubled, the peak operating frequency would drop from 6 GHz to 3.5 GHz. Half power beamwidth at the proposed test frequencies are shown for 100 MHz, 1 GHz, and 6 GHz. Deployment sequence of a typical triangular element is shown in Figure 5-15. Grouped similar to parachute material, the Chromel-R tricot weave mesh will feed from flexible retaining rings as the truss elements straighten. Details of the mesh system can be seen in Figure 5-16. The white line mesh material is pulled tangentially down to four hexagonal flats in each triangular element by the webbing. Three tension lines (dotted lines) attached to the webbing and the adjustable screw jacks at the spider move the mesh to its best position. This system will be used during ground fabrication and for adjustment by the crew in space if necessary. A summary of the maximum distortion is shown in Table 5-1. The key element in mesh adjustment is the method and accuracy of the measurement system. In this study a modulated laser system (Figure 5-17) was developed capable of measuring the surface contour to ±0.006 in. from the feed point 40 ft above the mesh. A rotating mirror system would allow a complete scan of the reflector in 30 seconds. Test of the mesh reflectance has shown that local targets are not needed and a continual measurement of the mesh surface is possible.

## DEPLOYMENT SEQUENCE
### Reflective Mesh

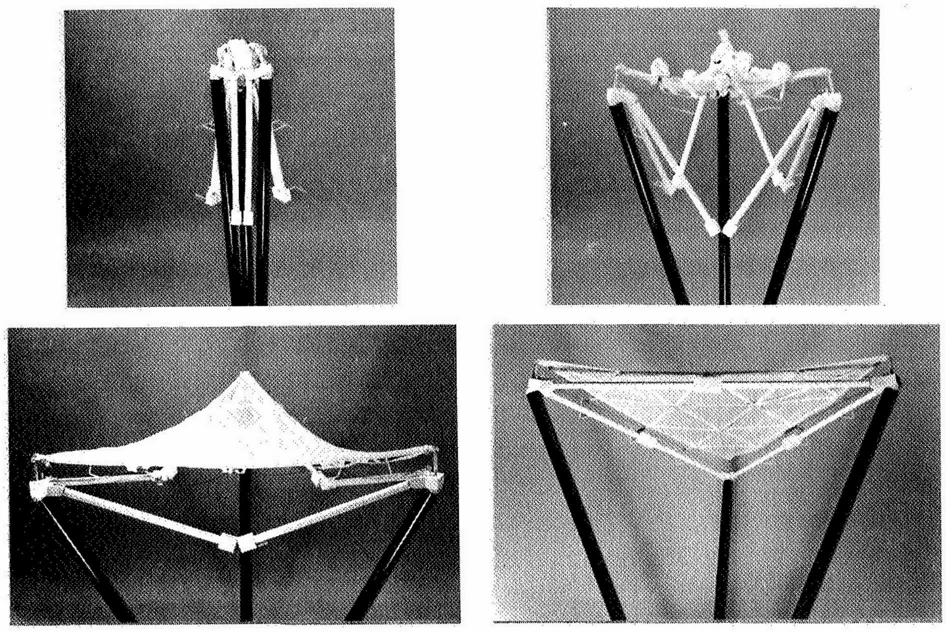

Figure 5-15

Table 5-1. Distortion Summary
100 Ft Diameter Paraboloid

| | |
|---|---|
| Tooling Location | 0.005 |
| Bearing Accumulation | 0.025 |
| Mesh Installation | 0.060 |
| Mesh Contour | 0.040 |
| Thermal (side) | 0.066 |
| Dynamic | 0.010 |
| | 0.206 in. |

RMS = 0.101 in.; $\sigma/D = 8.5 \times 10^{-5}$

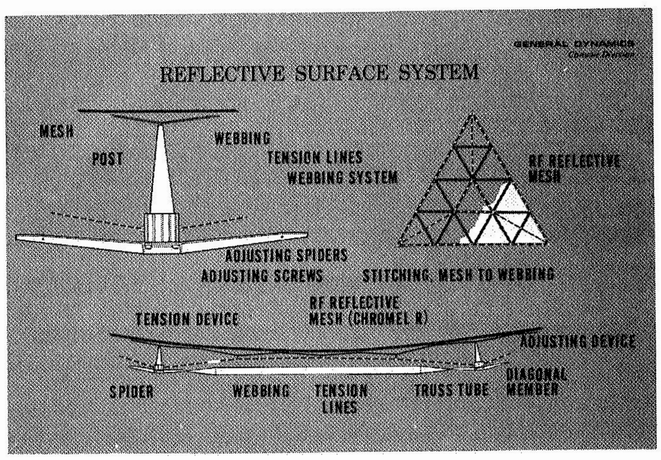


Figure 5-16

A printout of the surface contour, recommended adjustments, best fit paraboloid surface, and optimum focal point could be provided. Installed at the focal point, on the pressurized feed and electronic compartment, Figure 5-18, the laser measurement unit, monitored by the crew, would determine the optimum surface contour. Repeated measurements will be made during EVA adjustments, Figure 5-19, and confirmation of the best fit focal point. The feed cone will then be swung back over the laser and boresighted through three corner reflectors to match the optimum focal point, established by the laser.

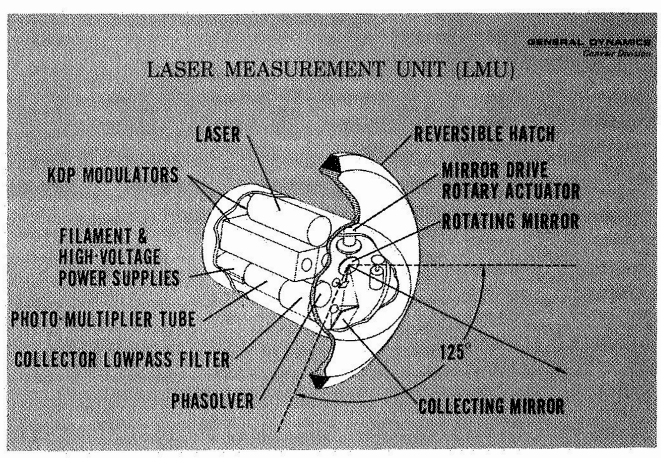


Figure 5-17

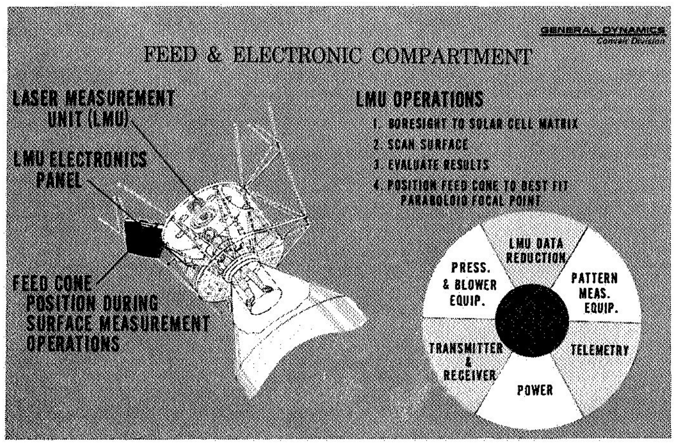


Figure 5-18

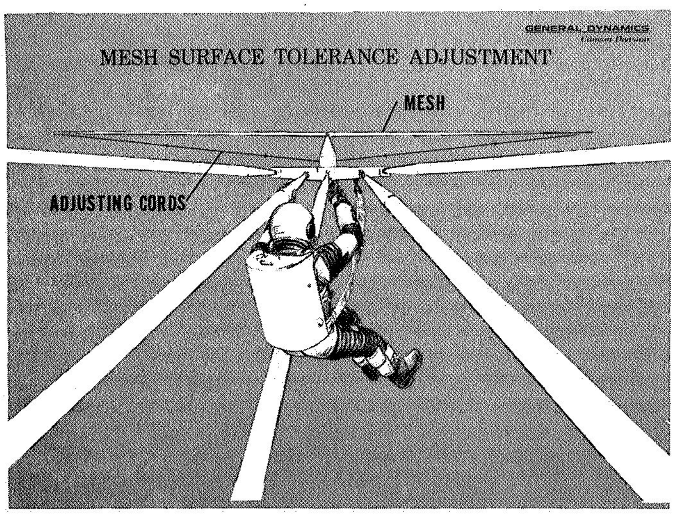


Figure 5-19

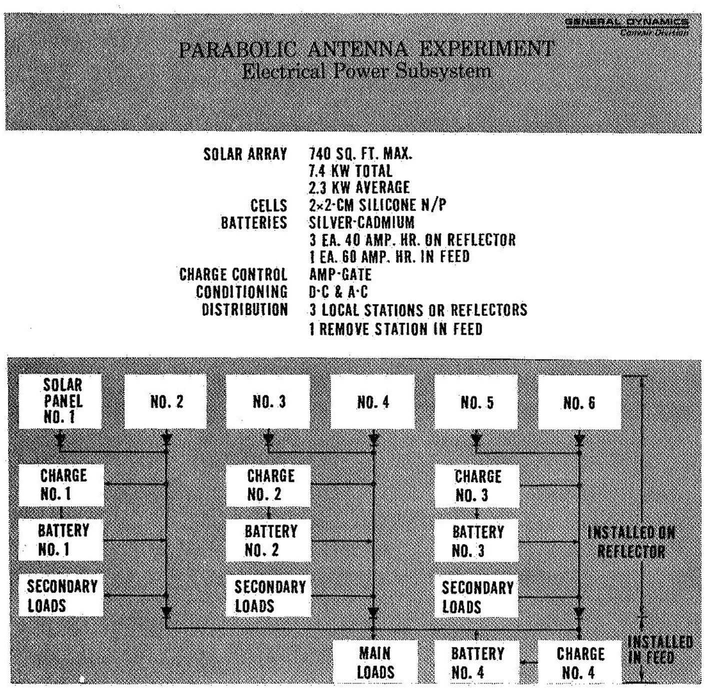


Figure 5-20

5.5.2 FEED AND ELECTRONIC COMPARTMENT. Beside the laser system, the eight-foot-diameter pressurized feed electronic compartment contains the transmitter and receiver, batteries, power conditioning and monitoring equipment, and the telemetry. With man required to perform, support, or monitor the experiment over the full 12-day period, a shirtsleeve environment is required in the compartment. An $O_2$ pressurization system compatible with the CSM is provided for the compartment. Environmental control comes from the CSM. Cold plates and external radiators will stabilize the electronic equipment heat load within the Apollo limits. All equipment must operate in either a pressurized or unpressurized environment since the compartment is used as an airlock during EVA tasks.

5.5.3 POWER SYSTEM. Power system is a major cost item in the experiment. As the program progresses every effort should be made to monitor and reduce the power requirements. Figure 5-20 outlines the basic elements of the system. No development items are anticipated.

5.5.4 ATTITUDE CONTROL SYSTEM. To provide the angular rate and attitude control limits required by the experiment (Figures 5-21 and 5-22) the hybrid control torque system of Figure 5-23 was configured. A low $I_{sp}$, but highly reliable, cold gas reaction

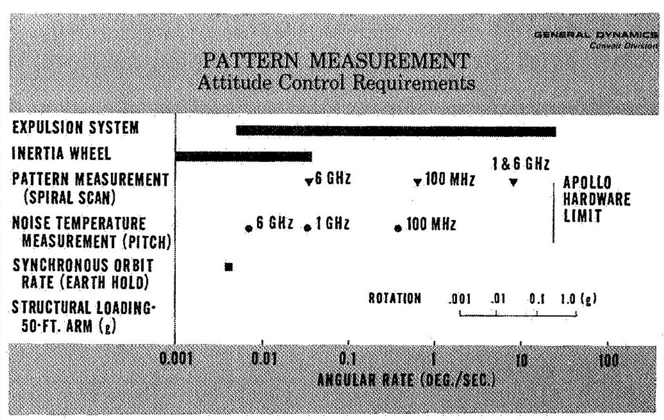


Figure 5-21

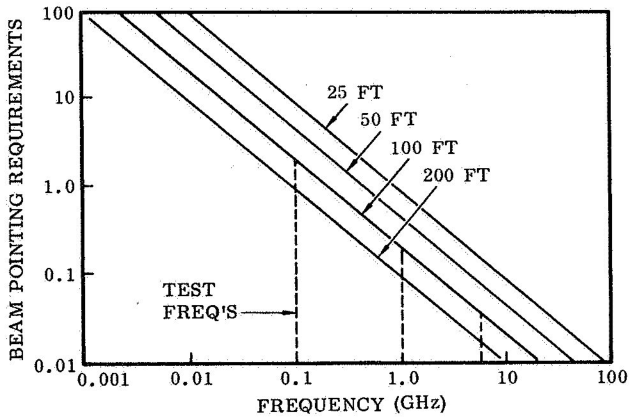


Figure 5-22

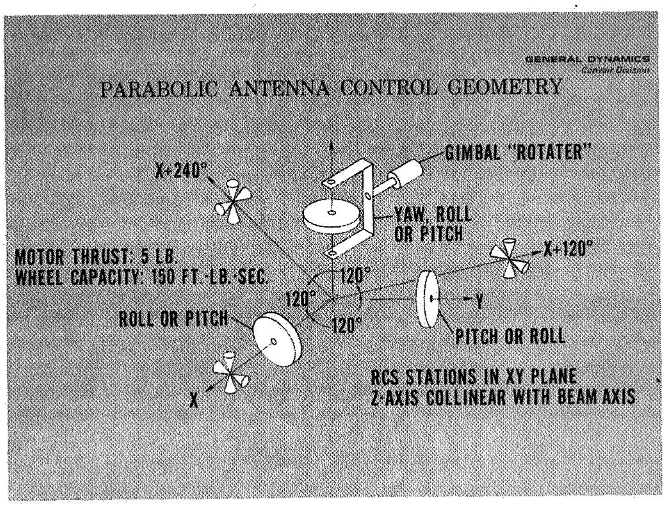


Figure 5-23

system is efficient when acting on the long moment arm that is permissible with the truss antenna. The expulsion system is used for corrections down to 0.06 deg and station keeping. A redundant inertia wheel system is used for correction from 0.06 deg to 0.001 deg. A two-year life system requires 200 pounds of propellant with 70 pounds assigned to station keeping.

The star tracker system illustrated in Figure 5-24 provides the desired sensor accuracy and references all pointing to the inertial reference of the star field. A horizon scanner and a solar aspect seeker will determine the basic reference with the star tracker providing the fine control. Precision in torque control and attitude sensing can be rendered useless if the working platform elastic properties effectively make the star tracker the analog of a micrometer on the end of an oscillating yardstick. Figure 5-25 effectively points up the nature of the deterioration in the ability to hold attitude as a function of a change in natural frequency (or stiffness) of the antenna structure. In this example a reduction from 1.6 cps to 1.0 cps will preclude pointing to within ±0.02 degree. The 100-ft diameter antenna with a 5-ft depth has a reflector natural frequency of 1.6 cps. A computer-aided analysis has been developed to determine the basic modes of the reflector.

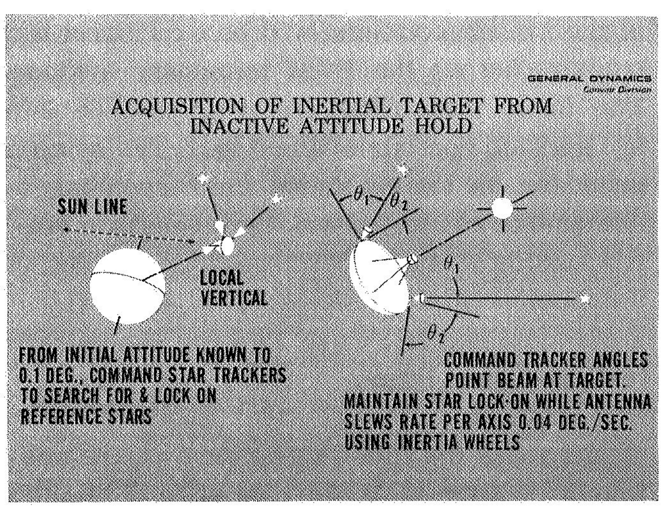


Figure 5-24

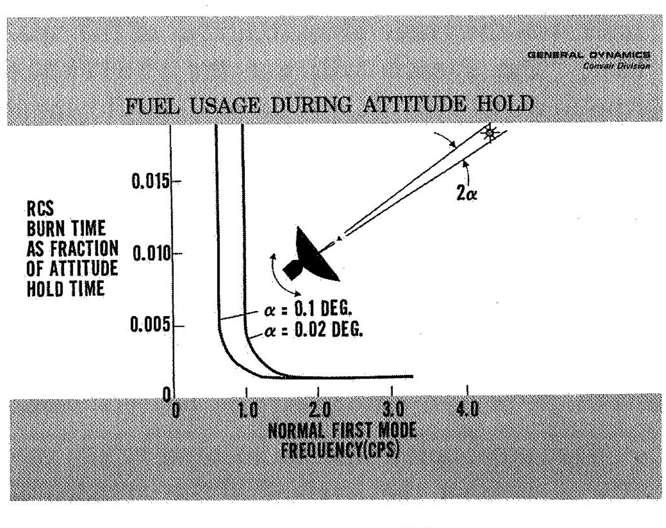


Figure 5-25

This ultimate dependence on structural stiffness can be met by the truss type parabolic antenna because of its inherent stiffness and the relative ease with which the stiffness can be increased to levels commensurate with the extremes of pointing accuracy envisioned for this experiment and the future system.

5.5.5 TELEMETRY AND COMMAND. The design of the telemetry and command system is based primarily on the requirements and constraints of performing unmanned intermittently for a one to five year period preceded by an initial 14 day manned operation. The data collected from the various measurements in the experiment will be channeled to the USB transponder for transmission to earth.

Two telemetry antenna systems will be available: high gain or omni. The space erectable antenna (52 db) can be used for a high rate data link when not used in pattern and gain measurement tests.

The command system is required so that a number of vehicle functions may be initiated from the ground and to transmit digital data to the spacecraft. Functional commands are associated with the deployment of the experiment, the control of antenna position, data storage, and transmission. Figure 5-26 illustrates the basic block diagram of the system. The system will use the available Apollo equipment and no new development areas are anticipated.

5.5.6 EXPERIMENT DATA CONDITIONING. along with the rf parameters, mechanical, thermal, and surface measurement data will be collected for antenna evaluation (Figure 5-27).

Thermocouples will be located at key positions across the surface of the antenna to evaluate thermal gradients. Strain measurements will be instrumented in near locations to correlate mechanical distortion and stresses with thermal gradients. Microswitches will monitor the lockup of the hinged joints in the antenna structure. The status of the switch contacts will be scanned and transmitted via the PCM telemetry system. The laser measuring unit will provide data for surface contour evaluation. The system produces an analog signal corresponding to distance or range and two digital signals corresponding to angles. These data will define the antenna surface shape. A similar set of measurements is required for the pattern test discussed later.

Bit rates for this experiment are one-fourth the normal Apollo rate of 51.2 kilobits per second.

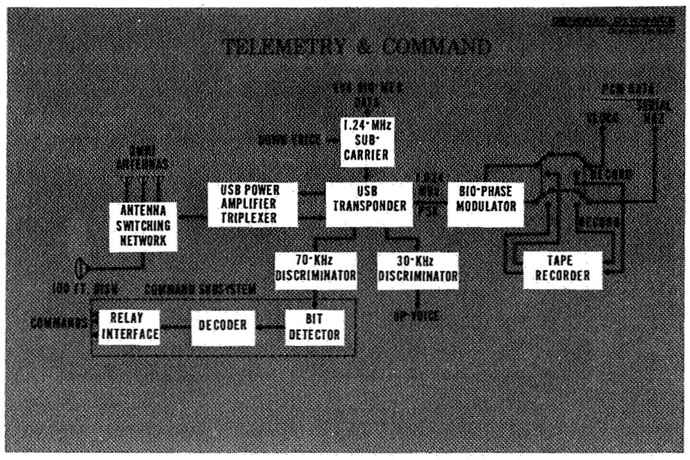


Figure 5-26

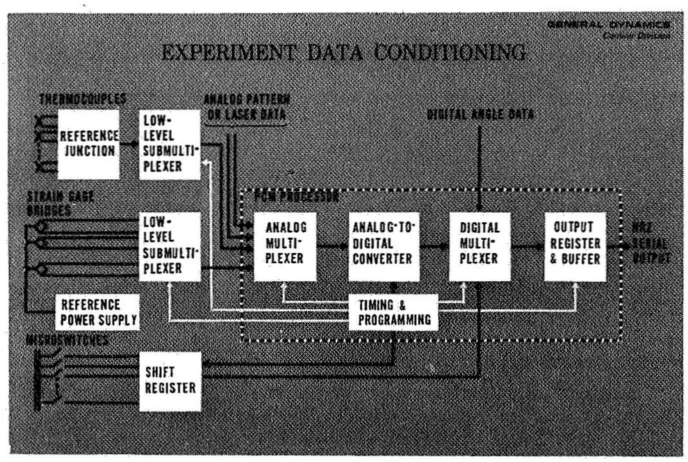


Figure 5-27

## 5.6 PATTERN MEASUREMENT

Radiation pattern and absolute gain define the basic characteristics of an antenna. The measurement of the pattern consists of determining the relative antenna radiation intensity and the direction at which the intensity occurs. Two orthogonal polarization components of relative power density or electric field intensity, and two corresponding angles in orthogonal components, are measured at each pattern point to describe the pattern completely. Figure 5-28 shows a basic field intensity measuring and recording system to perform the test.

In the proposed measurement diagram, patterns of both space erectable and standard gain antennas are mapped simultaneously to determine the two pattern peaks. Both receiving systems are calibrated from a common signal generator so that the differential effects of transmission line components and receiver operation can be accounted for in calculations of the absolute gain.

5.6.1 ORBIT AND SOURCE SELECTION. Pattern measurements with large antennas having corresponding small beamwidths are best performed at synchronous orbit. In synchronous orbit there is continuous ground source visibility, data transfer and processing is facilitated, orbital motion can be utilized

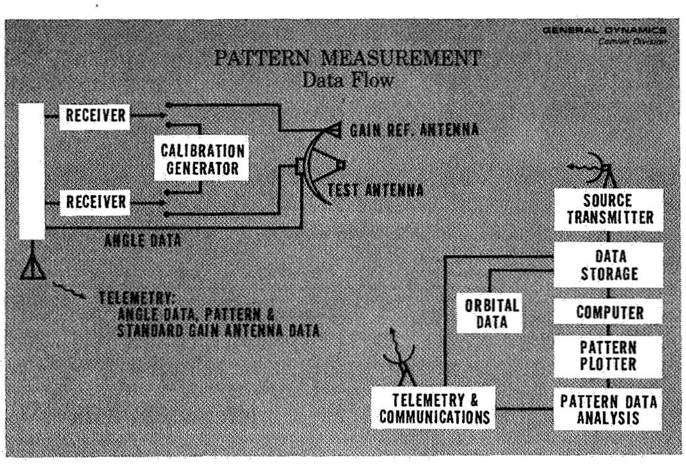


Figure 5-28

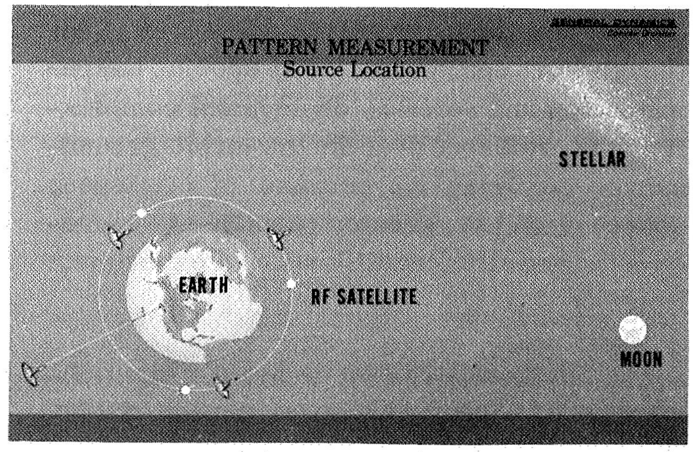


Figure 5-29

for one pattern coordinate angular change, orbital rates are sufficiently low to allow use of low torque auxiliary attitude control units, and steering maneuvers are simplified. Furthermore, the antenna is evaluated in the most likely future system operational orbit.

Severe measurement limitations are encountered in the low orbit configurations when attempting to use a ground source. Operational time is limited to approximately 20 minutes per day (two consecutive passes in a 24-hour period with 10 minutes over target). Within these time constraints, only limited pattern measurements can be conducted generating approximately 70 percent of the significant data required for complete evaluation of antenna performance.

Of the various radiating sources considered, Figure 5-29, (earth-based, stellar, rf satellite, lunar-based), a ground station was selected because of the advantages of experiment flexibility, polarization and radiated power control, frequency control, extensive instrumentation facilities, environmental control, and easy maintenance.

If the experiment is forced to a lower orbit (300 miles) by booster cost, availability, or other operational constraints, testing is best accomplished with an rf satellite source deployed from the parent spacecraft. Although more costly in terms of special instrumentation and satellite deployment complexity, it is preferred over the low-orbit-ground-source configuration because of continuous source visibility, greater quantity of recorded data, and efficient utilization of experiment time.

5.6.2 TRANSMITTER POWER REQUIREMENTS — SYNCHRONOUS ORBIT TEST. Figure 5-30 shows the ground transmitting source output power requirements for measuring the 100-ft parabolic antenna as a function of frequency for three common sizes of ground antennas in current use (30-, 60-, and 85-ft diameters). Assumptions are a -100 dbm receiver sensitivity, 2 db system losses, and a 40 db dynamic range for the pattern recording. Since the satellite, at the 120-degree longitude neutral stable point, would be within view of the JPL Goldstone facility, the power requirements with a 210-ft diameter ground antenna are also presented in Figure 5-30. None of the power requirements shown are difficult to achieve if the transmitters are located on the ground; any of these antennas could be used with suitable feed modifications to accommodate the test frequencies.

5.6.3 PATTERN SCANNING MANEUVERS. Considering the restrictions of mission time, data rates, hardware cost, propellant, and reaction control engine life, the most promising scanning techniques for space application are great circle and spiral scans (Figure 5-31).

The conical cut cannot be taken in space because of orbital motion. Scanning by a raster motion is eliminated by excessive fuel requirements. A potential variation of the raster scan, which conserves propellant, is a nodding or oscillating scan generated by a soft spring interconnection between a large mass (such as the CSM) and test antenna.

GROUND POWER REQUIRED
FOR PATTERN TESTS

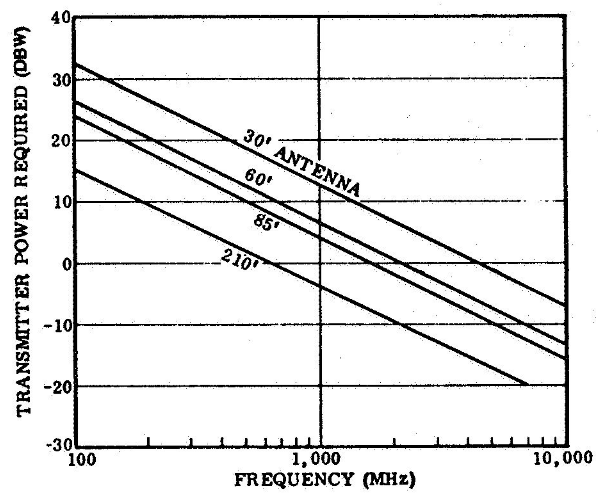


Figure 5-30

SCANNING TECHNIQUES FOR
PATTERN DATA GENERATION

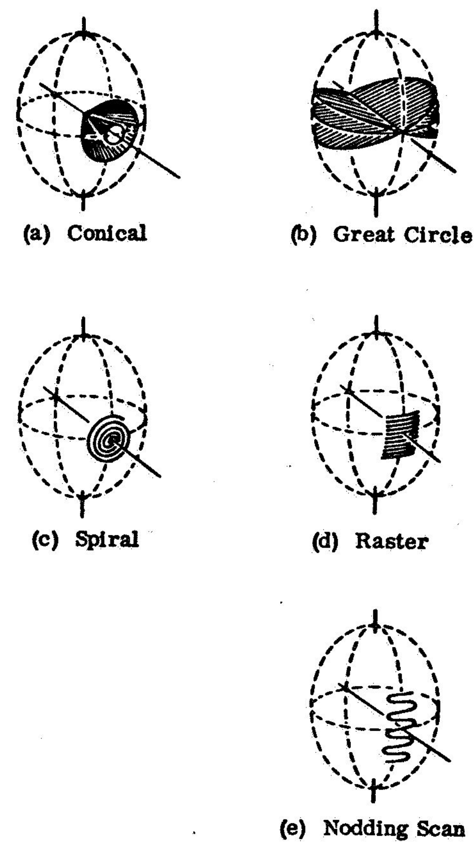


Figure 5-31

The system becomes a mechanical oscillator when excited, developing a sinusoidal trace over the radiation sphere.

Angular changes for pattern recordings are generated by two effects: orbital motion and spacecraft attitude changes induced by control devices. An independent attitude control system is designed into the antenna experiment utilizing cold gas engines and an inertia wheel system. The angular momentum units are effective attitude control devices for pattern mapping at synchronous altitudes where steering rates are small. They are used in this experiment to generate a fine spiral trace about the antenna axis (±1.5 deg) for boresight, focusing, and detailed mainlobe and sidelobe data. Combinations of RCS-induced spin rates and orbital motion are used

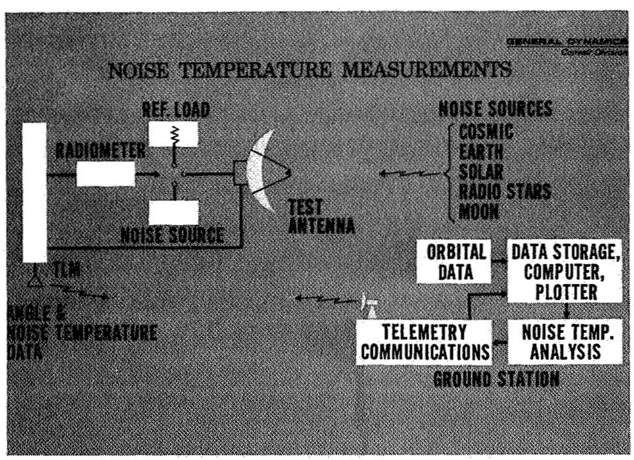


Figure 5-32

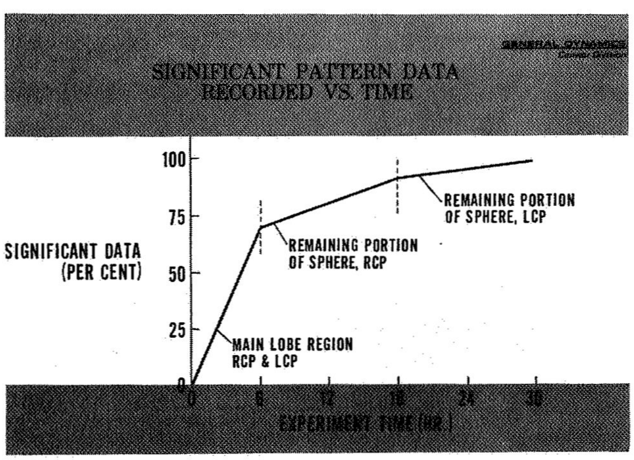


Figure 5-33

to generate a more coarse spiral trace over the radiation sphere. Angular momentum units are also used for most of the noise temperature measurements to create the slow angular drift rates required.

5.6.4 NOISE TEMPERATURE MEASUREMENTS. Noise temperature measurements require more sensitive instrumentation than the pattern measurements but are less taxing on the spacecraft attitude control system and measurement bit rate since slow drift rates and small sampling rates are involved. Figure 5-32 illustrates the noise temperature measurement system.

5.6.5 BIT RATE AND SIGNIFICANT DATA TIME. It is apparent that certain regions of the antenna radiation pattern have greater significance in the rf performance evaluation. Detailed pattern measurements over the main lobe and near-in-sidelobes will define the major portion of antenna characteristics. Since this is a small portion of the radiation sphere, mapping can be accomplished in a relatively short time. Second in importance is the remaining portion of the pattern containing wide-angle sidelobe, backlobe, and reference pattern data for long term antenna degradation analysis. This data, however, takes considerable time to record in detailed sampling increments. Least in importance but again requiring a large experiment time increment is the cross-polarized pattern component. The partial directivity contribution from this component can be obtained, with a fair degree of accuracy, from a scale model test.

Continuous measurement time required to complete a spiral scan over these regions is shown in Figure 5-33 for the 100-foot parabolic antenna at synchronous altitude.

Peak bit rate occurs when the 1 GHz and 6 GHz tests are simultaneously being performed, Table 5-2 outlines the measurements.

Table 5-2. Pattern and Gain Measurement Bit Rate (One RPM Rotational Rate)

| Measurement | Total Bits | Sampling Rate (samp./sec) | Bit Rate (b/s) |
|---|---|---|---|
| Two Angles | 28 | 150 | 4200 |
| Two Amplitudes (1 GHz) | 14 | 34 | 476 |
| Two Amplitudes (6 GHz) | 14 | 150 | 2100 |
| Time | | | 240 |
| Simultaneous Measurement Bit Rate, 1 and 6 GHz | | | 7016 |
| Measurement Bit Rate, 6 GHz Only | | | 6540 |

5.6.6 ASTRONAUT PARTICIPATION — RF MEASUREMENTS. Presence of the astronaut simplifies antenna measurement procedures, eliminates redundant equipment requirements, and enhances measurement reliability. His participation in pattern measurement experimental activity includes:

a. Calibration of electronic equipment by functional switches, amplitude control, and visual observation.

b. Alignment and periodic updating of the inertial measuring unit from celestial observations.

c. Checkout of rf instrumentation and spare unit installation.

d. Manual feed deployment if automated mechanism fails.

e. Make final feed adjustment based on boresight pattern data.

f. Monitoring data and log comments during experiment.

g. Boresighting optical trackers to mechanical pointing axis.

h. Maneuvering spacecraft to point at stationary and earth-based targets.

i. Aligning pointing axis to initial scan position and realigning periodically as discrete angular pattern sectors are covered.

j. Initiating and stopping scanning motions.

k. Aiding in target acquisition by monitoring amplitude of detected rf signals and by making visual observations.

l. Initiating alternate measurement procedures if failures occur.

m. Make final equipment checks and connections for unmanned mode of operation.

## 5.7 ASTRONAUT EQUIPMENT

To perform the tasks outlined in Figure 5-34 the astronaut will have a command console in the feed-electronic compartment. He will use the laser surface measurement system, evaluate the surface contour and recommended correction, and decide what action to take. He will steer the antenna to acquire the transmitter and monitor pattern measurements throughout the test. Aided by ground control he will decide which sections of the patterns test should be repeated. In the noise and pointing tests he will steer the

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SUMMARY OF ASTRONAUT TASKS

OBSERVE DEPLOYMENT PROCESS
INSPECTION - STRUCTURE, ELECTRONICS & POWER SYSTEMS
REPAIR - TUBES, MESH, LASER & ELECTRONICS
ADJUST TOLERANCE - MESH & FEED
PERFORM MESH CONTOUR TEST
PERFOPM PATTERN MEASUREMENT
PERFORM NOISE TEMPERATURE MEASUREMENT
ACQUIRE GROUND TARGETS
MAINTAIN STRUCTURE & EQUIPMENT
SET ANTENNA IN AUTOMATIC MODE
OBSERVE AUTOMATIC RESPONSE

Figure 5-34

antenna and initiate lockon to ground, satellite, and celestial targets. He will backcheck the attitude sensor equipment with position fixes at critical acquisition phases.

His potential EVA tools and spare parts are listed in Figure 5-35. A low reaction hand torque gun will be used to make mesh corrections. C clamps have broad application to positioning material to make adjustments or repairs. Wire and coaxial splicers are needed if deployment or boost vibration should damage the lines coming from the perpheral mounted equipment to the electronic compartment. Metal and mesh cutters could be used to release a damaged truss tube. Redundance of the truss will allow any one of the six elements in a hexagonal assembly to be removed without impairing the structural integrity of the antenna. A stadi optic could be used to approximate distances and make measurements of the truss and mesh. Each of the ACS engine modules can be removed and a new module plugged in. The laser system will have a spare. The laser mounts on a hatch in the electronic compartment and can be removed for repair or replacement. Power and coaxial line replacements may be stored on the side of the electronic compartment. A telescopic tubular element spare can be used to replace any damaged tubes or as a test of EVA dexterity in a simulated repair. Tests made on the mesh show that Velvco tape is ideal for joining a damaged area. If surfaces were marred during boost or deployment, the thermal coating would be repaired with either spray or tape during the EVA inspection. A lubrication would be useful for the tube and feed hinges and the adjustment system.

Simulation of the feed/electronic compartment and a EVA simulator on a hexagonal element of the reflector are needed to determine the feasibility of various crew tasks. Once the tasks are established, mockups will be used for training.

GENERAL DYNAMICS
Convair Division

ASTRONAUT TOOL & SPACE PARTS

| TOOLS | SPARES |
|---|---|
| HAND TORQUE | ATC PLUG-IN |
| C-CLAMPS | LASER MEASUREMENT UNIT |
| WIRE SPLICER | POWER LINE |
| COAX. SPLICER | COAX. LINE |
| VOLTMETER | TUBULAR ELEMENT 13 & 9 FT. (VARIABLE SIZE) |
| METAL & MESH CUTTER | VELCO TAPE (MESH) |
| POWER SAW | THERMAL COATING (SPRAY & TAPE) |
| WRENCH SOCKET | LUBRICATOR |
| SCREWDRIVER | |
| STADI CALIBRATED OPTIC | |

Figure 5-35

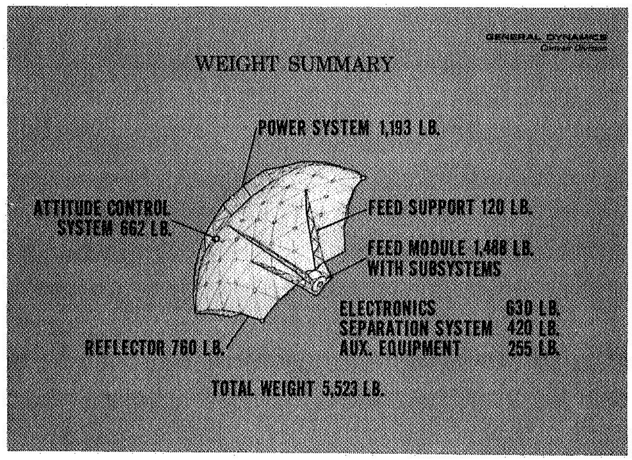


Figure 5-36

## 5.8 WEIGHT SUMMARY

Figure 5-36 contains the weight summary of a 100-ft-diameter antenna experiment. All systems are sized for a life of two years. Ample payload is available for a CSM/antenna launch with the Saturn V to synchronous orbit.

## 5.9 FABRICATION AND TEST

Figure 5-37 illustrates the proposed sequence. Four basic items — reflector, feed support, feed and electronic compartment, and the boost and transportation pallet — are fabricated. Upon completion of the basic truss of the reflector and feed, the antenna will be assembled in the packaged condition.

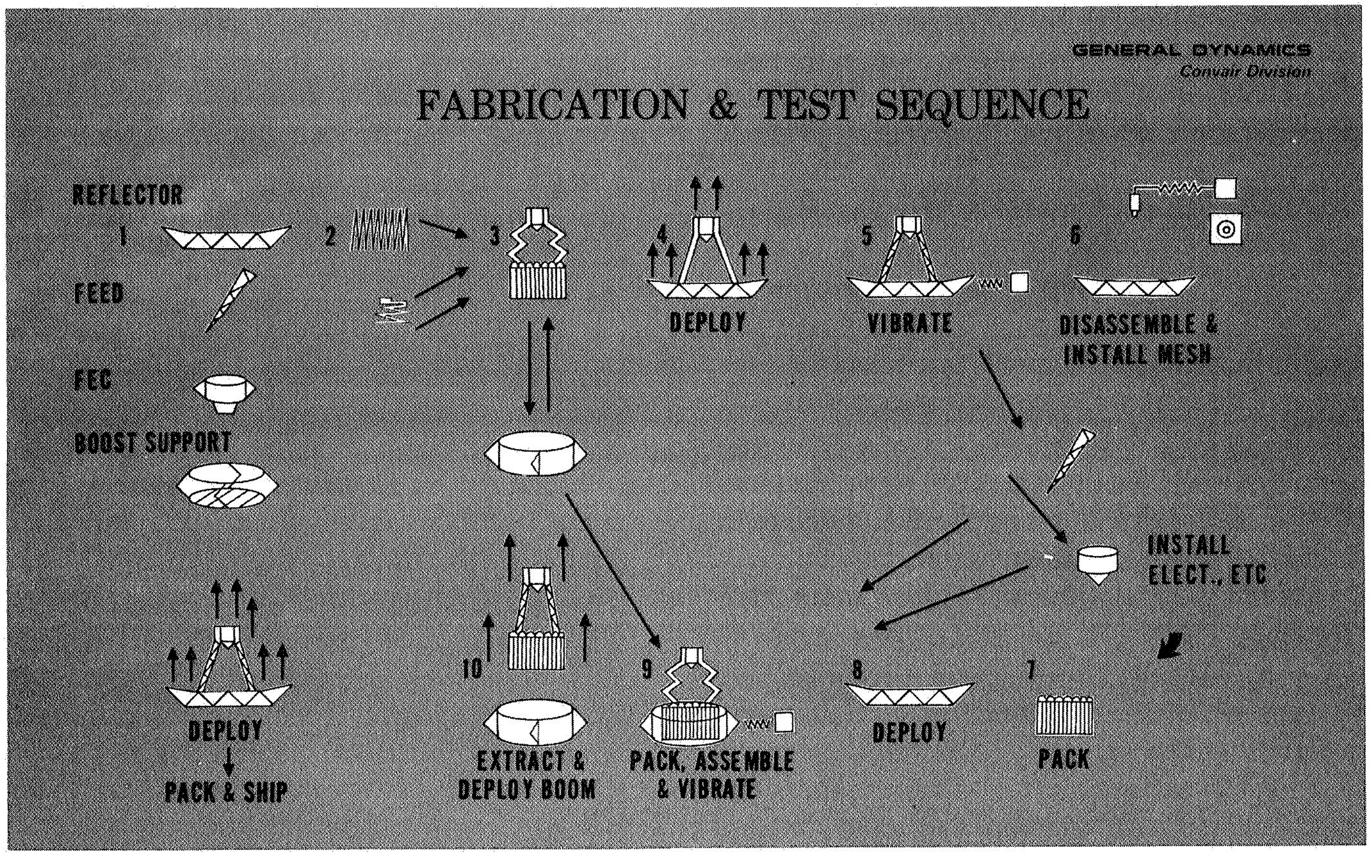


Figure 5-37

A fit check will be made in the pallet. A negator spring system will ride a trolley rail carrying the weight of each spider section to simulate zero gravity during deployment. The deployed antenna will then be vibrated to determine the primary modes. Disassembly and installation of the mesh on the reflector and equipment in the compartment follow. A separate packaging and deployment of the reflector will check out the system with the mesh on. The re-packaged complete assembly will then be vibrated to boost loading condition and the full flight sequence simulated. RF testing will be performed on a scale antenna. A 30 to 50 mile test range would be required for the full size antenna. A similar sequence of operations will be required on the flight antenna. Figure 5-38 illustrates the fabrication jig. The zero-g trolley system will be on a spider web network above the antenna. Temperature control of $\pm 2^{\circ}$F is desirable to prevent tolerance buildup.

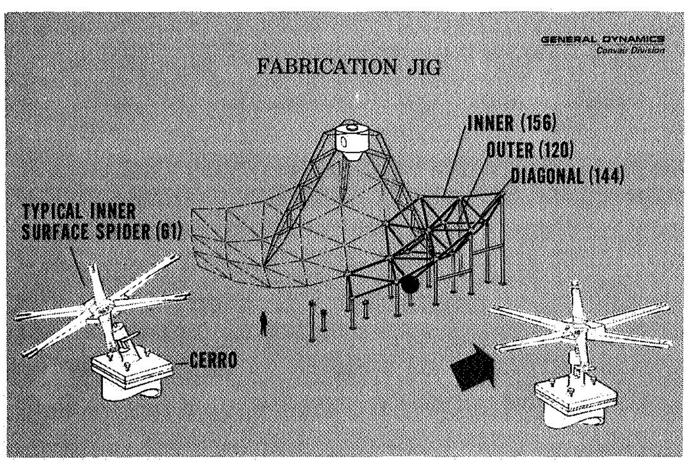


Figure 5-38

A summary of ground testing is shown in Figure 5-39.

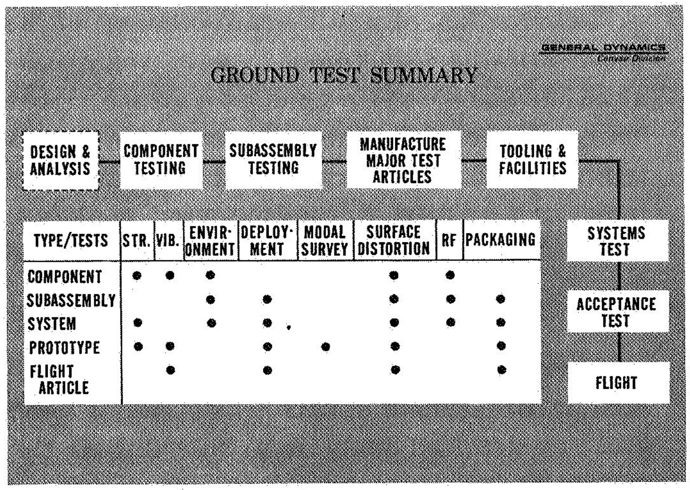


Figure 5-39

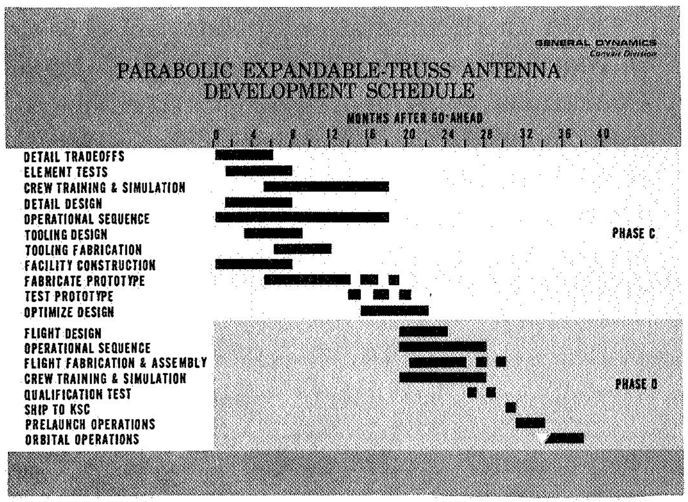


Figure 5-40

## 5.10 COST AND SCHEDULE

A schedule of the major tasks is given in Figure 5-40. Fabrication and test of the prototype will be followed by a short design optimization period before fabrication and qualification of the flight experiment. Prototype testing will be completed in 21 months. Launch of the experiment can take place 34 months from go-ahead.

Cost of the complete experiment, including development, ground and training equipment, prototype fabrication and test, and one flight article, would be \$27.3 million. Figure 5-41 outlines the fiscal funding requirements.

## 5.11 DEVELOPMENT TASKS

Primary development tasks that require more effort are itemized in Figure 5-42.

The laser measurement unit is within the state of art of laser technology but has not previously been assembled to perform the proposed measurement task. Development may be divided into three parts: laser system, scanning system, and software printout system. Figure 5-43 shows a tentative schedule.

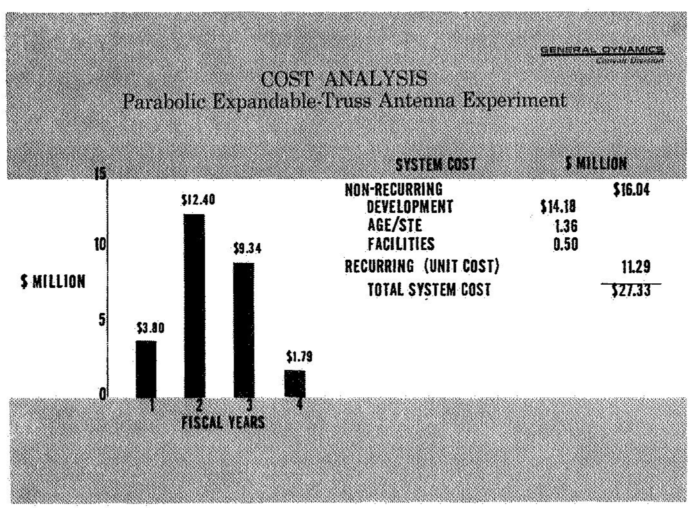


Figure 5-41

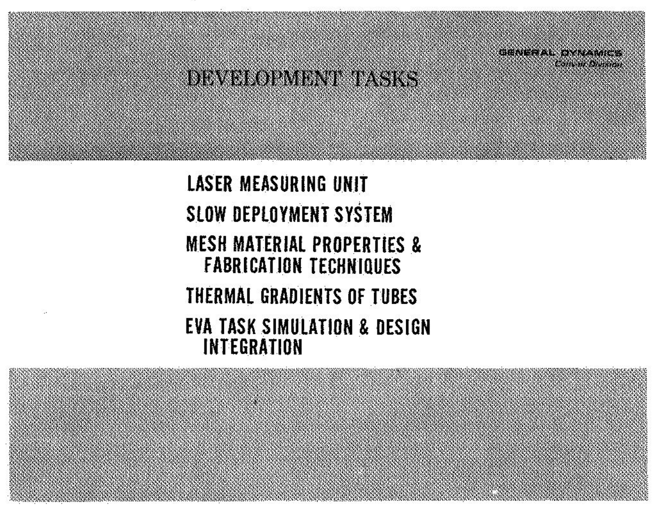


Figure 5-42

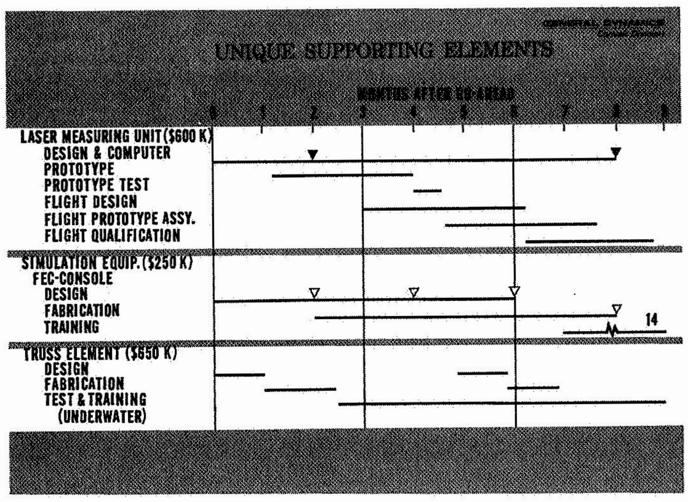


Figure 5-43

Slow deployment is feasible but details must be worked out. Many schemes of controlling deployment have been suggested. The best of these concepts must be evaluated and tests made on the available model to demonstrate the design.

Fabrication of a full scale element of the mesh with the adjustment system would substantiate the design concept and work out "bugs" before full scale fabrication is attempted. Thermal cycling in vacuum of the mesh is needed to test the mesh coatings. General physical properties also need to be verified at the operating environment.

Thermal gradients through the tubular structure and mesh at various sun angles need a more detailed analysis to substantiate the anticipated thermal gradients. Temperature distribution around perforated tubes should be tested in a vacuum chamber.

EVA task simulation in underwater tests is required to provide early information. Operations in the pressurized feed compartment must also be simulated to better determine required equipment and time task allotments. Figure 5-43 illustrates the development and training period for the crew simulators.